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EVALUATION OF ENVIRONMENTAL PROFILES FOR RELIABILITY DEMONSTRATION

Grumman Aerospace Corporation

EVALUATION OF ENVIRONMENTAL PROFILES
FOR RELIABILITY DEMONSTRATION

Rome Air Development Center
Air Force Systems Command
Griffiss Air Force Base, New York 13441

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This report describes the results of a 19-month study to determine the adequacy of the environmental profiles of MIL-STD-781, "Reliability Tests: Exponential Distribution," and provide recommendations for their improvement. The environmental and failure history of a sample of 95 avionic weapons replaceable assemblies (WRAs) served as the vehicle for this evaluation. Environmental data during demonstration testing and actual field exposure for each WRA was collected, analyzed, and compared. Test failure history and field failure experience were

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reviewed and reliability measures for each determined. Comparisons of lab-field reliability differences with environmental differences indicated that: insufficient thermal cycling; inadequate vibration testing in terms of level, duration and frequency; and lack of moisture exposures were deficiencies in the current environmental profiles. Recommended profiles which would eliminate these weaknesses are presented. In addition, the need for consistent ground rules, failure scoring criteria, and a strong end-item burn-in test are described.

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PREFACE

This Final Technical Report was prepared by Grumman Aerospace Corporation, Reliability, Maintainability and Safety Section, Bethpage, New York, under Contract F30602-73-C-0317, Job Order 55190262, for Rome Air Development Center, Griffiss Air Force Base, New York. Mr. Lester J. Gubbins (RBRS) was the RADC Project Engineer.

The effort described was accomplished during the period June 1973 through January 1975.

Technical consultation on environmental analyses was provided by Mr. David L. Earls (AFFDL/FEE) of the Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio.

In addition to Messrs. Hirschberger and Dantowitz, other Grumman study team members were: Mr. John Campbell, Mr. Lawrence Flanagan, Mr. John Fletcher, Mr. Harvey Hott, Mr. Lawrence Kutin, Miss Linda Kuttner, Mr. Neil Marcus, Mr. Joseph Osche, Mr. Joseph Popolo, Mr. Herbert Quartin, and Mr. Thomas Sexton.

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Introduction

1.1 Background

For years both government and industry have observed that the reliability of avionic equipment in the field, has not generally attained the level demonstrated in the laboratory. Studies predict and laboratory demonstrations verify the achievement of satisfactory levels of reliability, yet once in the field, equipments fail at a rate significantly higher than expected. Observed differences have in many cases been estimated to be at least a factor of three to as much as an order of magnitude.

The specific reasons for such variances have been the subject of different government funded and industry sponsored studies. It is generally accepted that the main contributors to these apparently inconsistent statistics are certain obvious and other more obscure operational considerations. These factors can usually be grouped however, as differences: attributable to the equipment itself; between the environments encountered during test and end item usage; and in the derived data base upon which the equipment reliability is determined. These factors and the probable reasons for the contradictory estimates are summarized in Table 1.

ENVIRONMENT

Data and experience seem to indicate that the difference between laboratory tests and field environmental exposure is one of the more significant reasons for avionic estimated reliability incompatibility. Increased environmental stress levels on hardware due to modern high speed, high performance aircraft are responsible for many field failures. A study performed by Grumman (ref. 1) clearly indicates that almost 50% of the field failures of the equipments studied were environmentally related.

Almost without exception, laboratory demonstration tests of subsystems composed of several constituent black boxes, are performed in one test facility and at one level of stress. In the aircraft these same units may be located in different areas and therefore potentially exposed to different environmental conditions. Even if all items are designated to be the same

TABLE 1 FACTORS CONTRIBUTING TO RELIABILITY DIFFERENCES

FACTOR	POSSIBLE REASON
ENVIRONMENT	• Limited test exposure in terms of type, duration and level. • Variability of in-field site conditions. • Dissimilarity between test conditions & operational requirements. • Test environment is usually controlled & benign. Constrained by test equipment capability & flexibility. • Potential for mishandling greater in the field.
EQUIPMENT	• Test article not necessarily representative in terms of parts, processes or materials. • Maturity & configuration evolution between test specimen & later production units. • Sophistication of peripheral test equipment to identify & localize problems. • Accessibility for diagnostic & corrective maintenance controlled during testing.
DATA	• Failure definition & classification of ground rules may not be consistent. • Accuracy & completeness is a function of personnel skill & motivation. • Estimation techniques inconsistencies. • Equipment traceability & discernment of secondary problems very limited or nonexistent.

MIL-E-5400 "class" (ref. 2), which indicates similar thermal and altitude design and test requirements only, other environmental conditions (notably vibration) may vary considerably.

Small, light items are not likely to incur much damage due to handling. As equipment weight and/or volume increases, the damage potential, due to banging, dragging, etc., may increase significantly. In the laboratory, the reliability test article is generally given the 'kid glove' treatment, thereby minimizing the probability of handling damage. However, in the field, because of aircraft installation and removal requirements, this effect may become significantly more pronounced and logically would increase with item weight and volume.

It is apparent therefore that reliability demonstration test environments do not sufficiently or adequately reflect field usage.

EQUIPMENT

Quite often, in order to meet contractual commitments, production equipment is delivered before the reliability demonstration test is complete. One of the reasons that the reliability of field hardware may be lower than that demonstrated during testing is that many problems, detected during the demonstration test for which a change will not be incorporated into the production hardware until some later date, will still be present in the initially delivered units.

During the various phases of a program, equipments produced during the pre-production phase are reworked for certain purposes. If a reliability demonstration test is to be conducted, one or more of these 'early' units will usually be designated as the reliability sample(s). Experience indicates that the earmarked unit(s) is often assembled, controlled and inspected with more care than the average production hardware. In addition, these test units may see more operating time prior to actual test, than is accrued on each production unit before delivery, resulting in a biased test specimen.

The conditions under which scheduled and unscheduled maintenance activities are performed on the equipment, in the laboratory as well as in the

field, are quite different and create the very real potential for introducing undesireable, but possibly unavoidable, contributors to equipment unreliability. The skill level of personnel involved in failure diagnostics and repair can significantly affect the identification and classification of problems, which eventually become data elements. Inadequate equipment handling practices due to lack of proper maintenance stands, tools and procedures or personnel motivation could result in induced failures, hence erroneous conclusions. As packaging density increases, this situation would be more pronounced in the field than in the laboratory since the demonstration tests are generally conducted with highly skilled personnel.

High quality level parts (TX, ER, etc.) are so designated because they undergo more rigorous inspection and testing procedures than other parts which are physically and functionally identical. In the field these parts should be better able to endure long environmental exposures than the low quality level types. Since they are more environmentally tolerant, MTBF values for equipments with a high percentage of high reliability parts may therefore be more closely related to laboratory demonstration values than units with a much lower percentage of high reliability parts.

During laboratory demonstration tests, because of practical considerations or expediency, not all functions (i.e., performance parameters) may be monitored. Out-of-tolerance situations and even certain 'hard' failures may occur during environmental exposures and never be detected during the test or during final acceptance test. In an aircraft, however, these anomalies could very likely appear and correctly be counted as a failure.

DATA

Ground rules for failure definition and time measurement must be consistent to assure that field and laboratory reliability comparisons are valid. It is obvious that a large disparity between these parameters, when used in defining MTBF and scoring failures, will result in a significant difference in reliability.

During the earlier stages of aircraft deployment, test equipment and hardware may not be completely compatible and this incompatibility could result in erroneous failure diagnosis during troubleshooting. Incorrect

failure data would then be entered into the data recording system.

The specified value of θ_o, as defined in MIL-STD-781 (ref. 3), is the measure that must be demonstrated in the laboratory. This MTBF requirement is established by the user of the equipment and must be satisfied by the equipment manufacturer by means of an estimate which is based on the results of the demonstration test. It is therefore extremely important that the prediction be realistic based on the best available data. In addition, it is also important that a uniform prediction policy be maintained for all avionic equipment. Current techniques utilize data from a wide variety of sources, often resulting in equipment whose measured value of MTBF does not approach the original requirement.

1.2 SCOPE

Although all of the indicated considerations have, to varying degrees, contributed to the noncorrelation of field and laboratory demonstrated reliability, the scope of this study has been deliberately focused on the effects of the environmental factors. Certain of the other contributors were also investigated when the field data indicated a significant impact.

The environmental factors appeared to be the most fruitful area of investigation because:

- experience has shown that the reliability of equipments is significantly affected by the environmental stresses to which they are exposed.
- the laboratory test environments, which have remained basically unchanged over the years, no longer represent the induced environments of today's high performance aircraft.
- laboratory test conditions (e.g., special handling equipment, controlled "clean room" environments, well defined exposure durations, etc.) do not reflect what the equipment will experience in the end usage

1.3 Objectives

The principal objectives of this study were:

- Determine the adequacy of the environmental profiles of MIL-STD-781 in simulating field stresses.
- Where inadequacies exist, provide recommended new test profiles for inclusion in MIL-STD-781.

Secondary objectives included:

- Determine the adequacy and provide recommendations for improving demonstration test ground rules and scoring criteria.
- Identify changes needed in reliability prediction methods to produce better correlation between demonstration test and field results.

1.4 Approach

The following interrelated activities (summarized in Figure 1) were performed to achieve the above objectives.

- Line Removal Units (LRU) were selected in accordance with a set of criteria developed to assure a cross section of function, location, cooling and mounting provisions. Ninety-five distinct LRU's were selected and the physical characteristics and operational requirements of each LRU was compiled. Note that "LRU" is a generic term used to describe any replaceable package of an avionic equipment or system as installed in an aircraft weapon system. The equivalent Navy term is Weapons Replaceable Assembly (WRA). Inasmuch as the data analyzed for this study pertained to WRA's, that particular acronym will be used throughout the report.
- The level and duration of all environmental exposures during demonstration tests and in the field were determined. This included the compilation and review of all environmental data extracted from test plans and reports, available flight instrumentation data (e.g. References 26-30) and engineering analyses. The results of all pertinent, previous studies were included in the determination. The source documentation was also reviewed for consistency of ground rules, assumptions, failure criteria, corrective action requirements and effectivity.

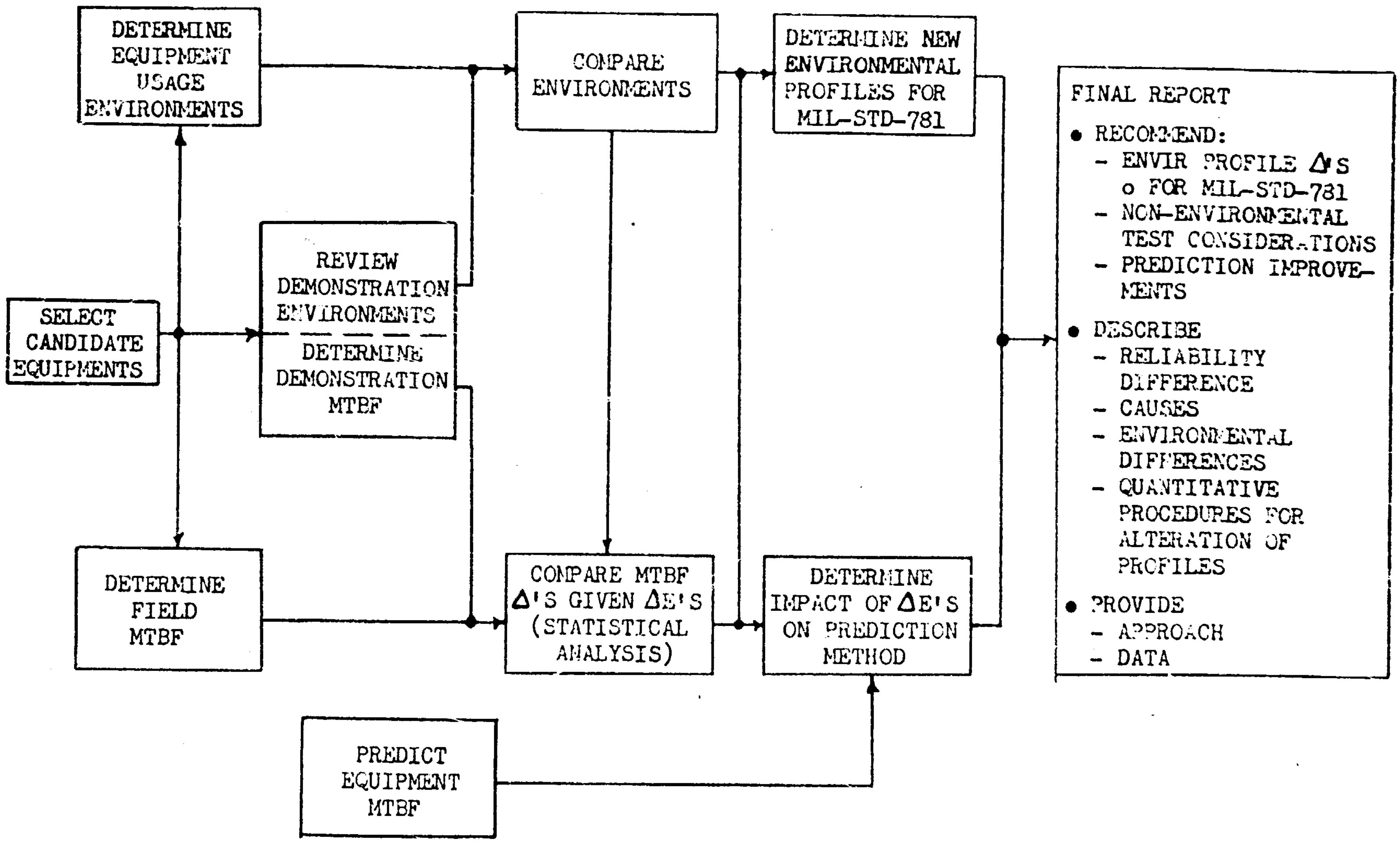


FIGURE 1 EVALUATION OF ENVIRONMENTAL PROFILES FOR RELIABILITY DEMONSTRATION STUDY FLOW

- The results of the demonstration tests and field usage were reviewed in terms of experienced failures, test time, number of flight hours, etc., to determine the demonstration and field MTBF's. This necessitated the establishment of and the adherence to certain prediction groundrules and assumptions. Failure narratives contained in test reports and operational performance data as contained in the Navy Maintenance, Material, and Management (3M) system were reviewed for applicability to this study. A reprediction of the reliability of each WRA, based upon part populations, electrical and thermal stresses and the failure rates of the coordination copy of MIL-HDBK-217B (ref. 23) was also accomplished.
- The outputs of the Environmental Analysis and MTBF Analysis were studied through the use of statistical analysis techniques. Significant causitive environmental factors that related to the difference in reliability between the laboratory and the field were identified.
- Candidate demonstration test profiles including environments, exposure time, sequencing, levels, etc., were developed by using the results of the statistical analysis, previous study results, and constraints imposed by equipment design, test equipment, and testing economics.

Section II

Equipment Selection/Data Requirements

2.1 Selection Criteria

In order to satisfy both the primary and secondary objectives of this study, it was necessary to formulate a hierarchy of selection criteria for choosing the equipments to be analyzed. These criteria described either a design or utilization characteristic of the candidate equipment or the availability of data. Except for those which pertained to data availability, the selection discriminators had to have a direct bearing on the major issues of the study, that is, they had to:

- potentially affect an equipment's environment
- potentially affect the reliability of the equipment

The overriding consideration in the assessment of each criterion was to provide a comprehensive equipment blend in terms of design complexity and environmental exposure. Certain general conclusions and recommendations pertaining to environmental demonstration testing of all types of avionic equipment could then be formulated from this sample of analyzed equipments. The items were chosen on the basis of satisfying as many of the criteria as possible. In addition, since it was highly improbable that all of the items typified a representative sample of each selection screen, a criteria precedence was established. But here again, the utmost concern was to maintain a good cross-sectional character to the group of analyzed items. These criteria in rank order included:

- Laboratory Demonstration - In order to perform any evaluation, it was implicit in the basic study objective that each selected equipment must have been subjected to a reliability demonstration test. The equipments chosen were all subjected to such tests, primarily levels E & F of MIL-STD-781 (ref. 3). Certain equipments were purposely selected because they had not been exposed to a MIL-STD-781 profile, but had been tested to environments representing typical mission profiles.

- Field Data - Availability of data and a thorough knowledge of environmental "use" parameters also dictated the choice of hardware. Since the equipments selected are presently included in the Navy inventory, field data was available through the 3M system and was readily supplemented by available preflight and flight test data. Further, environmental profiles had been determined for each of the aircraft selected.

- Weapon System Diversity - Since the environmental exposure an equipment experiences is dependent upon the characteristics of the aircraft in which it is installed and its associated mission, equipments have been selected from a variety of aircraft weapon systems to assure a cross section of environmental exposures. The aircraft represented in this selection have included the following physical and functional considerations:

  - Turboprop, turbojet and turbofan propulsion
  - Subsonic and supersonic speeds
  - Land and carrier based aircraft
  - Attack, fighter, and surveillance missions

- Cooling Method and Mounting - In order to establish a base for comparison of thermal cycling effects, equipment was selected from classes 1A, 1X, 2, 2A and 2X as defined by MIL-E-5400 (ref. 2) and encompassed natural convection, liquid, forced air, and fan cooling methods. In a similar manner, both hard-mounted and isolator-mounted equipments were selected to enable an assessment of different field performance since MIL-STD-781 tests call for hardmounting of all test hardware.

- Location in Aircraft - To provide diversity of environmental intensity and to establish a base for MIL-STD-781 test comparisons (since laboratory tests are primarily conducted on subsystems under one set of environments while actual aircraft location may include many environments for the same subsystem), equipment was selected

whose WRA's were located in the nose, fuselage body, cockpit, tail, etc.

- Usage - All equipment selected and used in the aircraft types defined were produced in quantity. Further, all common equipment functions were represented in the selected hardware, i.e., radar, communications, navigation, computer functions, displays, A-D converters, high-power transmission, sensors, etc.

- Contemporary Design - The primary driver used when considering vintage of equipment was state-of-the-art technology. Since the study output was aimed at recommending profiles for future tests, it was important that analysis be performed on equipment representing current technology. Further, if the equipment evaluated was too old, maximum reliability growth would have been achieved via corrective/design improvement action, and any comparison between laboratory and field would contain a built-in bias. Of course, equipment had to be mature enough to be deployed so that data would be available. This contradiction was mitigated somewhat by selecting equipment with a large complement of microcircuitry (I/C's, etc.) as well as other unique and current design features. In addition, equipment was also selected which included a mix of parts/quality levels.

2.2 Equipment Descriptions

Based upon the criteria indicated above, the items to be analyzed during the study were selected. It was originally decided to analyze 15 separate equipments from four different aircraft. One of the first major decisions in conducting this study was to expand this group to provide a wider spectrum of equipment technology and environmental experience. This was accomplished by considering each WRA comprising the 15 subsystems as a discrete study equipment. Each WRA was successfully tested against the selection criteria and were considered valid study subjects in that the cross-sectional character of the group was preserved. An additional benefit in going to the WRA level was to broaden the data aggregate since in general, a particular equipment or system was environmentally tested under identical and simultaneous conditions. In actual use, however, each WRA was not co-located

with any of the other items in the system, thereby introducing its own in situ environmental exposure. The end result was that ninety five WRA's were selected for analysis. A description of each of these is contained in Appendix A.

Table 2 is a summary of the more significant physical and design characteristics of those chosen.

Abbreviations have been used in the columns of Table 2 to define various cooling and mounting methods. The following legend describes these:

COOLING METHOD

A	=	Ambient - Convective cooling - No supplemental air
FA	=	Forced Air - Supplemental cooling from environmental control system air passes directly over components
FA-O	=	Forced Air and Oil - Forced air (as above) plus oil-to-air heat exchanger in cold plate
CP	=	Cold Plate - Supplemental cooling from environmental control system air passes through cold plate (conductive cooling), not over components
IF	=	Internal Fan - Integral WRA fan draws in ambient air

Note that in the succeeding analyses, the WRA's that are ambient or internal fan cooled have been grouped together and are referred to as "ambient cooled" WRA's. Similarly, the WRA's which are forced air, forced air and oil, or cold plate cooled have been grouped together and are referred to as "forced air cooled" WRA's.

MOUNTING METHOD

H	=	Hard
IS	=	Isolator Mounted

2.3 Data Requirements

One of the major activities during this study was the accumulation, analysis and interpretation of data from many diverse sources. The data is categorically of a design, environmental and reliability nature.

TABLE 2 - PHYSICAL AND DESIGN CHARACTERISTICS OF SELECTED WRA's

WRA NO.	FUNCTION	VINTAGE (YEAR OF FINAL DESIGN)	MIL-E-5400 CLASS	COOLING METHOD	MOUNTING METHOD	WEIGHT (POUNDS)	POWER DISSIPATED (WATTS)	VOLUME (CU. IN.)	PIECE PARTS (EXCLUDING MIS-CELLANEOUS HARD-WARE)	PACKAGING DENSITY (PARTS/CU. IN.)	% MICROCIRCUITS
1	RF-REC.-XMTR	1966	2X	FA	IS	28.0	51	1320	1531	1.16	7.64
2	RF-REC.-XMTR	1966	2X	FA-O	H	247.4	7681	7064	1906	0.26	0
3	RF-REC.-XMTR	1970	2	A	H	200	233	7200	2875	0.40	7.37
4	RF-REC.-XMTR	1966	2X	FA	IS	30.0	55	1380	1834	1.33	5.94
5	RF-REC.-XMTR	1968	2	A	H	37.4	50	2413	4338	1.8	2.72
6	RF-REC.-XMTR	1966	2X	FA	IS	36.0	77	1540	2469	1.6	8.34
7	RF-REC.-XMTR	1970	2X	CP	IS	70	620	2002	643	0.32	2.02
8	RF-REC.-XMTR	1966	2X	FA-O	H	244.4	8194	7064	1806	0.26	0
9	RF-REC.-XMTR	1972	1X	FA	IS	22.5	83	721	3307	4.59	5.5
10	RF-REC.-XMTR	1966	2X	FA	IS	34.0	76	1540	2381	1.55	8.65
11	SIG. PROCESS.	1970	2	A	H	1.09	-	48	491	10.23	0
12	SIG. PROCESS.	1970	2	A	H	1.87	-	80	238	2.98	0
13	SIG. PROCESS.	1970	2X	CP	H	32.0	90	1693	3549	2.1	12.43
14	SIG. PROCESS.	1970	1X	FA	H	30.0	137	1731	882	0.51	61.22
15	SIG. PROCESS.	1966	2	A	H	1.6	15	71	51	0.72	0
16	SIG. PROCESS.	1966	2	A	H	9.0	15	308	14	0.05	0
17	SIG. PROCESS.	1966	2X	FA	IS	30.0	69	1729	1292	0.75	15.33
18	SIG. PROCESS.	1966	2	A	H	2.2	15	76	33	0.43	0
19	SIG. PROCESS.	1966	2X	FA	H	63.0	99	3623	989	0.27	4.65
20	SIG. PROCESS.	1968	2	IF	H	31.0	149	1122	1820	1.6	41.81
21	SIG. PROCESS.	1972	1	A	IS	7.9	18	250	633	2.53	3.32
22	SIG. PROCESS.	1968	2	A	H	13.0	25	248	843	3.4	3.2

TABLE 2 - PHYSICAL AND DESIGN CHARACTERISTICS OF SELECTED WRA's (Continued)

WRA NO.	FUNCTION	VINTAGE (YEAR OF FINAL DESIGN)	MIL-E-5400 CLASS	COOLING METHOD	MOUNTING METHOD	WEIGHT (POUNDS)	POWER DISSIPATED (WATTS)	VOLUME (CU. IN.)	PIECE PARTS (EXCLUDING MIS-CELLANEOUS HARD-WARE)	PACKAGING DENSITY (PARTS/CU. IN.)	% MICROCIRCUITS
23	INTERFACE	1967	1X	FA	IS	53.0	408	2640	2906	1.1	65.11
24	INTERFACE	1967	2X	CP	IS	69	100	4896	2259	0.46	11.51
25	INTERFACE	1966	2	A	H	12.0	12.0	819	304	0.98	13.43
26	INTERFACE	1967	1X	FA	IS	36.0	164	2640	1016	0.39	39.37
27	INTERFACE	1969	2X	CP	H	44	125	2550	1537	0.60	42.29
28	INTERFACE	1966	2X	FA	IS	45.0	145	2665	975	0.37	63.59
29	INTERFACE	1972	1X	FA	IS	28.0	81	939	2048	2.18	23.39
30	INTERFACE	1970	2X	CP	H	19.0	40	1054	629	0.60	26.39
31	INTERFACE	1966	2	A	H	13.0	16.0	819	1200	1.47	6.5
32	INTERFACE	1967	1X	FA	IS	34.0	160	2640	1719	0.65	58.29
33	INTERFACE	1972	1	IF	IS	37.5	150	314	359	1.14	20.89
34	INTERFACE	1967	1X	FA	IS	4.5	-	288	392	1.4	95.92
35	INTERFACE	1966	2	A	H	15.5	70	638	303	0.48	0
36	INTERFACE	1972	1	A	IS	7.4	15	465	497	1.03	16.3
37	COMPUTER	1965	2X	CP	H	48.0	245	1595	4390	2.75	36.0
38	COMPUTER	1970	2	A	H	8.4	41	121	2010	16.61	15.52
39	COMPUTER	1970	2	A	H	8.2	41	121	2363	19.53	13.84
40	COMPUTER	1970	2	A	H	8.3	44	121	1797	14.85	12.3
41	COMPUTER	1967	1Y	FA	IS	4.5	33	208	519	2.5	96.72
42	COMPUTER	1967	1X	FA	IS	4.5	33	208	444	2.2	95.72
43	COMPUTER	1967	1Y	FA	IS	4.5	33	208	450	2.2	92.67

TABLE 2 - PHYSICAL AND DESIGN CHARACTERISTICS OF SELECTED WRA's (Continued)

WRA NO.	FUNCTION	VINTAGE (YEAR OF FINAL DESIGN)	MIL-E-5400 CLASS	COOLING METHOD	MOUNTING METHOD	WEIGHT (POUNDS)	POWER DISSIPATED (WATTS)	VOLUME (CU. IN.)	PIECE PARTS (EXCLUDING MISCELLANEOUS HARDWARE)	PACKAGING DENSITY (PARTS/CU. IN.)	% MICROCIRCUITS
44	COMPUTER	1968	2	IF	H	31.5	138	2271	5823	2.6	8.88
45	COMPUTER	1970	1X	FA	H	29.0	137	1742	755	0.43	66.89
46	COMPUTER	1970	2X	FA	H	37.4	250	1165	2311	1.98	28.78
47	COMPUTER	1967	1X	FA	IS	15.0	-	620	1427	2.3	15.0
48	COMPUTER	1970	2X	FA	IS	30.5	245	1013	1851	1.83	50.73
49	COMPUTER	1970	1X	FA	H	34.0	167	1741	701	0.40	53.32
50	COMPUTER	1966	2X	FA	IS	47.5	560	4158	6678	1.61	40.76
51	COMPUTER	1969	1	A	IS	20.0	90	504	1345	2.68	25.8
52	COMPUTER	1967	1X	FA	IS	23.0	132	1550	778	0.50	16.58
53	DISPLAY & CONTR.	1970	2	A	H	3.3	30	-	31	-	0
54	DISPLAY & CONTR.	1966	2	A	H	10.0	58	589	628	1.07	28.34
55	DISPLAY & CONTR.	1969	2	A	H	22.0	36	1121	233	0.21	0
56	DISPLAY & CONTR.	1966	2	A	H	2.0	5	140	45	0.32	0
57	DISPLAY & CONTR.	1970	2	A	H	3.9	28	28	28	1.0	7.14
58	DISPLAY & CONTR.	1966	2	A	H	5.0	7	290	63	0.22	0
59	DISPLAY & CONTR.	1968	2	A	H	58.0	65	2485	1963	0.79	11.46
60	DISPLAY & CONTR.	1966	2	A	H	4.0	9	357	56	0.16	3.57
61	DISPLAY & CONTR.	1970	2	IF	H	16.5	150	672	634	0.94	5.52
62	DISPLAY & CONTR.	1966	2	A	H	3.4	3	247	59	0.24	0
63	DISPLAY & CONTR.	1972	1	A	H	2.8	30	130	133	1.0	0
64	DISPLAY & CONTR.	1966	2	A	H	1.0	8	83	17	0.21	0

TABLE 2 - PHYSICAL AND DESIGN CHARACTERISTICS OF SELECTED WRA's (Continued)

WRA NO.	FUNCTION	VINTAGE (YEAR OF FINAL DESIGN)	MIL-E-5400 CLASS	COOLING METHOD	MOUNTING METHOD	WEIGHT (POUNDS)	POWER DISSIPATED (WATTS)	VOLUME (CU. IN.)	PIECE PARTS (EXCLUDING MIS-CELLANEOUS HARD-WARE)	PACKAGING DENSITY (PARTS/CU. IN.)	% MICROCIRCUITS
65	DISPLAY & CONTR.	1970	2	IF	H	16.3	150	672	549	0.82	9.29
66	DISPLAY & CONTR.	1968	2	A	H	3.4	15	135	605	4.5	0
67	DISPLAY & CONTR.	1966	2	A	H	40.0	169	3257	790	0.24	21.9
68	DISPLAY & CONTR.	1970	1A	IF	H	36	121	2246	535	0.24	21.31
69	DISPLAY & CONTR.	1972	1	A	H	6.0	25	130	842	6.48	3.56
70	DISPLAY & CONTR.	1966	2	A	H	17.0	65	8721	1041	0.12	20.9
71	DISPLAY & CONTR.	1970	2	A	H	13.8	31.5	1147	1460	1.27	37.19
72	DISPLAY & CONTR.	1966	2	A	H	7.0	53	486	150	0.31	0
73	DISPLAY & CONTR.	1970	1A	IF	H	40.0	220	1953	591	0.30	2.2
74	DISPLAY & CONTR.	1966	2	A	H	25.0	74	1530	447	0.29	2.46
75	DISPLAY & CONTR.	1970	1A	IF	H	38.0	270	1866	565	0.30	3.36
76	DISPLAY & CONTR.	1967	1X	FA	IS	16.0	15	2240	616	0.28	11.69
77	DISPLAY & CONTR.	1966	2	A	H	3.0	67	197	55	0.28	0
78	POWER SUPL. & SW.	1967	1X	FA	IS	22.0	96	1548	148	0.10	0
79	POWER SUPL. & SW.	1966	2X	FA	IS	52.0	680	1370	655	0.48	0.31
80	POWER SUPL. & SW.	1970	2	A	H	20.5	102	986	938	0.95	7.14
81	POWER SUPL. & SW.	1967	1X	FA	IS	7	127	480	481	1.0	0
82	POWER SUPL. & SW.	1970	2X	CP	H	36.0	265	1002	750	0.75	1.2
83	POWER SUPL. & SW.	1966	2X	FA	IS	40.0	320	1370	237	0.17	1.27
84	POWER SUPL. & SW.	1972	1	A	IS	54.5	1540	1053	1748	1.66	3.89

TABLE 2 - PHYSICAL AND DESIGN CHARACTERISTICS OF SELECTED WRA's (Continued)

WRA NO.	FUNCTION	VINTAGE (YEAR OF FINAL DESIGN)	MIL-E-5400 CLASS	COOLING METHOD	MOUNTING METHOD	WEIGHT (POUNDS)	POWER DISSIPATED (WATTS)	VOLUME (CU. IN.)	PIECE PARTS (EXCLUDING MISCELLANEOUS HARDWARE)	PACKAGING DENSITY (PARTS/CU. IN.)	% MICROCIRCUITS
85	POWER SUPL. & SW.	1968	2	IF	H	7.3	64	156	144	0.92	2.78
86	POWER SUPL. & SW.	1967	1X	FA	IS	8.0	106	428	278	0.65	3.6
87	ELECTRO-MECH.	1970	2	A	H	1.72	41	39.9	65	2.13	7.06
88	ELECTRO-MECH.	1970	2	A	H	1.72	41	39.9	86	2.16	8.14
89	ELECTRO-MECH.	1970	2	A	H	1.72	41	39.9	143	3.58	6.29
90	ELECTRO-MECH.	1967	1X	FA	IS	35.0	85	1450	902	0.62	38.69
91	ELECTRO-MECH.	1970	2	A	H	0.77	30	[illegible]	3	0.75	0
92	RACK & CAB.	1972	1X	FA	IS	18.0	-	1090	4	0.004	0
93	RACK & CAB.	1967	1X	FA	IS	77.0	-	11,800	52	0.004	0
94	RACK & CAB.	1972	1X	FA	IS	24.7	-	1190	7	0.006	0
95	RACK & CAB.	1967	1X	FA	IS	161.0	-	25,700	45	0.002	0

All of the physical characteristics, i.e., weight, power, volume, internal power dissipation, etc., of the candidates were compiled. Part populations and electrical/mechanical stresses of the constituent parts were determined. The available Reliability Demonstration tests documentation which described the sequence, levels and duration of individual test programs were reviewed and the number of anomalies during these tests were recorded. A review of the 3M data system for 1973 indicated the in-field performance in terms of operating time and failures. In addition, all of the available development flight instrumentation data for the subject WRA's was compiled for which best engineering estimates were made of the natural and induced environmental conditions experienced during actual flight. All of this data was then assimilated in such a manner as to permit the determination of reliability measures and environmentally related differences. It was then possible to develop those design, test and/or environmental relationships that seemed to contribute most significantly to the reliability of the WRA's.

Section III

Environmental Analysis

3.1 Approach

This section describes the accumulation, analysis and summary of the environmental stresses that the selected equipments were exposed to during laboratory tests and in the field.

Wherever available, reliability demonstration test plans and reports were reviewed to determine the environmental levels to which the WRA's were exposed.

Field environmental data derived from actual measurements, analysis and actual experience as published in technical reports were compiled. In those cases, where actual flight measurements were not available, state-of-the-art analytical techniques were used to estimate the environmental levels at particular locations.

3.2 Demonstration Background

The Advisory Group on Reliability of Electronic Equipment (AGREE) was formed in 1952 by the Defense Department's Research and Development Board to "monitor and stimulate interest in reliability and recommend measures that would result in more reliable electronic equipment." In 1957, after two years of study, the project was completed and a report issued (Reference (4)). The results of Task Group 3 (there were nine groups formed) studies included recommendations for defining test parameters; accumulating data and estimating reliability figures of merit, based on the results of the test. The Group restricted environmental conditions to temperature,vibration, on-off cycling and input voltage cycling. Other conditions such as humidity, altitude and shock were purposely omitted in the belief that vibration and temperature exposures would reveal any marginal equipment design, sensitive to the untested environments. Current information indicates that the premise was incorrect and that failure modes due to humidity or altitude exposure are not duplicated by vibration/temperature.

The environments originally selected by Task Group 3 were chosen for the following reasons:

"Temperature: The temperature test is intended to approximate the service conditions under which the equipment will be required to operate."

"Vibration: This is not intended to be the most severe condition encountered but is felt adequate to show up workmanship items such as loose solder joints, loose parts such as screws, bits of wire, etc. This test is to be performed with the equipment mounted solidly on the vibration table without shock mounts."

"On-Off Cycling: This test is primarily to give the equipment a temperature cycle, causing the entire equipment to 'breathe,' expand and contract, be exposed to the surges of starting electrical power, plus checking actual operation."

"Input Voltage: Varying the input voltage both above and below the normal rated voltage places a strain on the various circuits and, since this is a normal condition in service, will reveal many weak conditions."

Table 3 summarizes the stress levels of the environmental conditions recommended by the Group.

This represented the beginning of reliability demonstration testing and the requirements were incorporated into specification MIL-R-26667 (USAF) (Reference (5)), followed by MIL-R-23094 (WEP) (Reference (6)), and finally, the MIL-STD-781 series currently in effect.

While the thermal level prescribed may be adequate, the vibration requirement and the lack of other environments (e.g., humidity, during the test period),are unrealistic based on data and studies performed by Grumman Aerospace Corp., the Air Force Flight Dynamics Laboratory at Wright-Patterson Air Force Base and others. Certainly the vibration test prescribed did not duplicate the field environment for jet aircraft (Reference (7)). In addition, the constraint of testing at one non-resonant frequency immediately precluded the detection of problems at other frequencies (Reference (8)).

As previously indicated, the temperature, on-off cycling and power tests were the only actual mission-related tests performed. Thus, while the reliability demonstration test was to be applied as part of the

TABLE 3 AGREE ENVIRONMENTS

STRESS LEVEL / ENVIRONMENTAL CONDITIONS	L (LIGHT)	M (MEDIUM)	H (HIGH)	X (EXTREME)
Temperature	25° ± 5 °C (68°F to 86°F)	40° ± 5°C (95°F to 113°F)	(Chamber) -54°C to +55°C (-65°F to 130°F)	-65°C to +71°C (-85°F to 160°F)
Vibration	None	25± 5 cps at 2g	Same as M	Same as M
On-Off Cycling	3 hrs. "on" plus long enough to stabilize at both high and low temp. by actual measurement.	Same as L	Same as L	Same as L
Input Voltage	Nominal	Max. specified permissible Voltage +0 - 2%	Same as M	Same as M

preproduction design qualification, the vibration test was only accomplished as a workmanship screen. It was not deliberately structured to verify any design adequacy. The potential deficiency of the test, however, was the exclusion of certain hostile but actual environments. The performance of the equipment during or after exposure to altitude, high humidity conditions, a variation of the input power frequency, etc., were not demonstrated. Based upon Grumman's experience and the field data examined during this study, these conditions (either singly or in combination) did eventually cause problems and were therefore considered viable areas of investigation for this study.

3.3 Demonstration Environments

Each of the equipments selected for this study had been subjected to a reliability demonstration test. The majority of the units were tested to the environmental profiles of MIL-STD-781 or MIL-R-23094 (WEP), which included only thermal cycling, on/off power cycling and fixed frequency vibration (see Figure 2). No voltage cycling was performed during any of the reliability demonstration tests conducted on the selected equipment. Input voltages were maintained within the equipment specification limits of +5% and -2% of nominal during all of the tests conducted. Power was shut off periodically during each cycle (per MIL-STD-781 requirements) and then reapplied but no attempt was made to run at the MIL-STD-781 limits of nominal, 90% or 110% of nominal. Certain items, however, were subjected to unique environmental profiles, which included humidity, shock, cooling air flow variations, etc., which were intended to simulate operational conditions. An example of this variation is presented in Figures 3 through 5.

Figure 3 depicts the test cycle (A) that was applied to ambient cooled equipment.

Figure 4 presents a similar cycle (A) applied to forced air cooled equipment. Note that in the latter case, the cooling air temperature was varied as well as the air flow. Figure 5 is a composite cycle which includes vibration, shock and a second (or "B") cycle which applied to some equipments (both ambient and forced air cooled equipment). This "B" cycle was performed under benign laboratory thermal ambient environments plus cooling air at a fixed rate. Each "A" cycle was followed by nine "B" cycles.

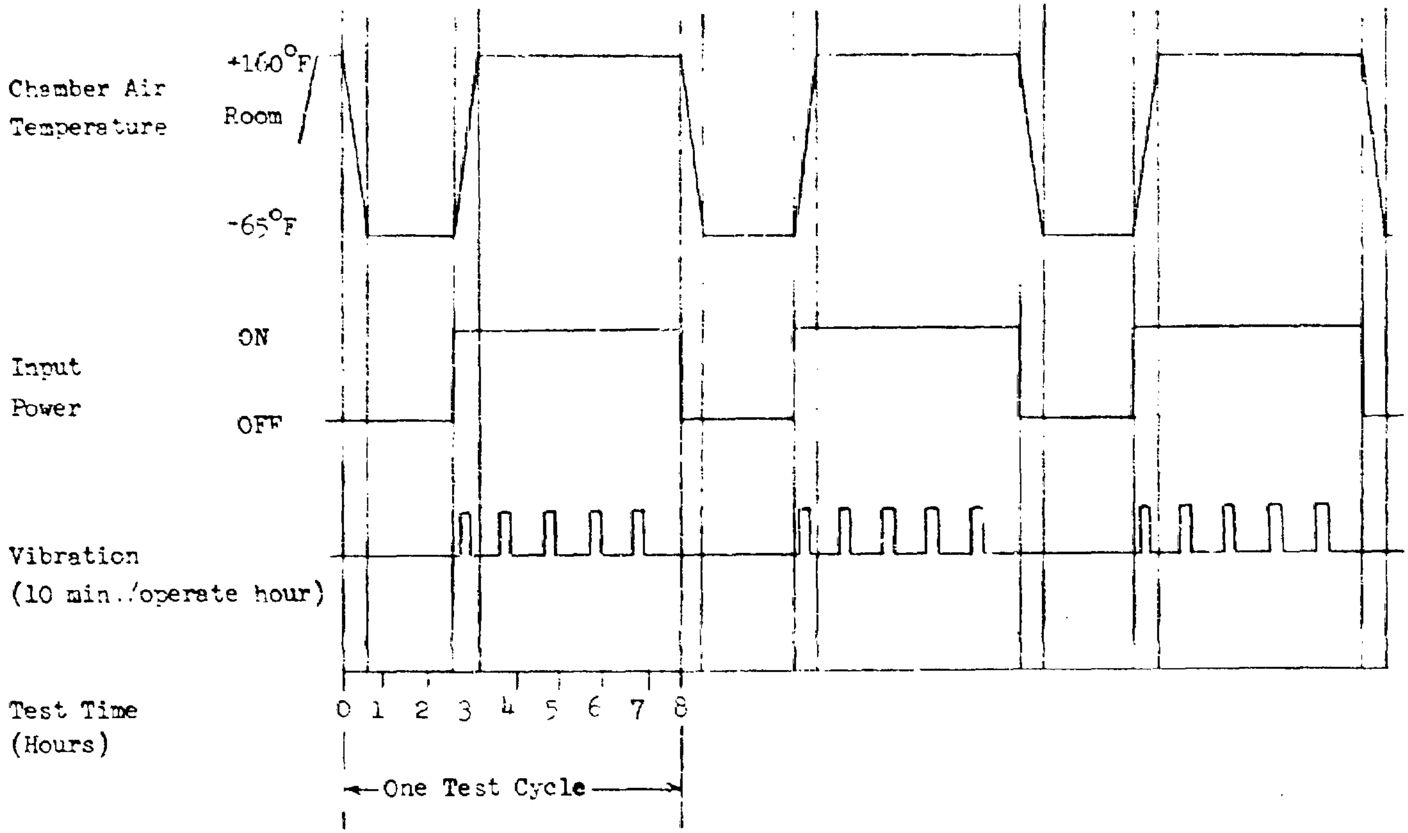


FIGURE 2 TYPICAL MIL-STD-781 PROFILE

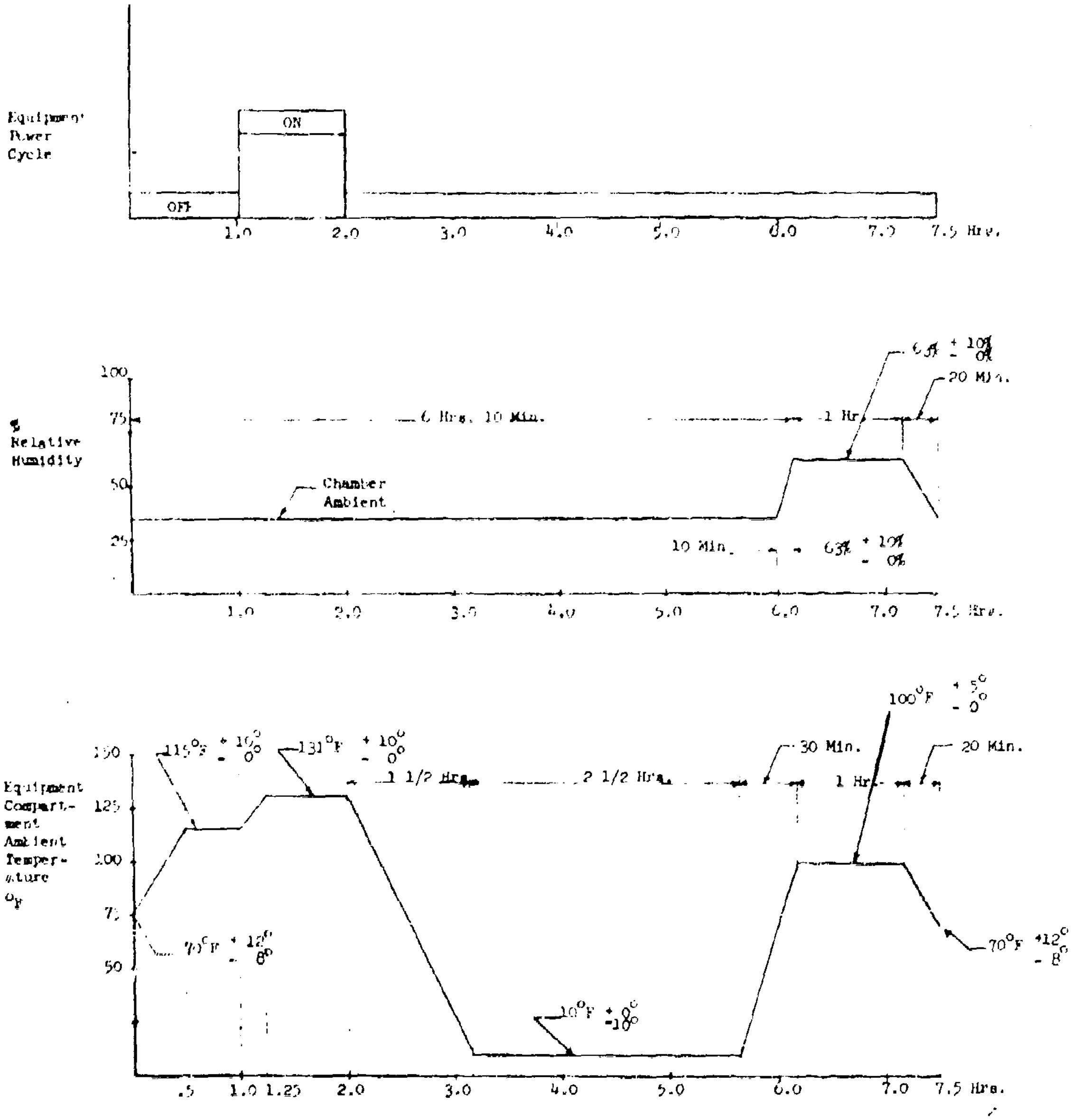


FIGURE 3 TEST CYCLE A - AMBIENT COOLED EQUIPMENT

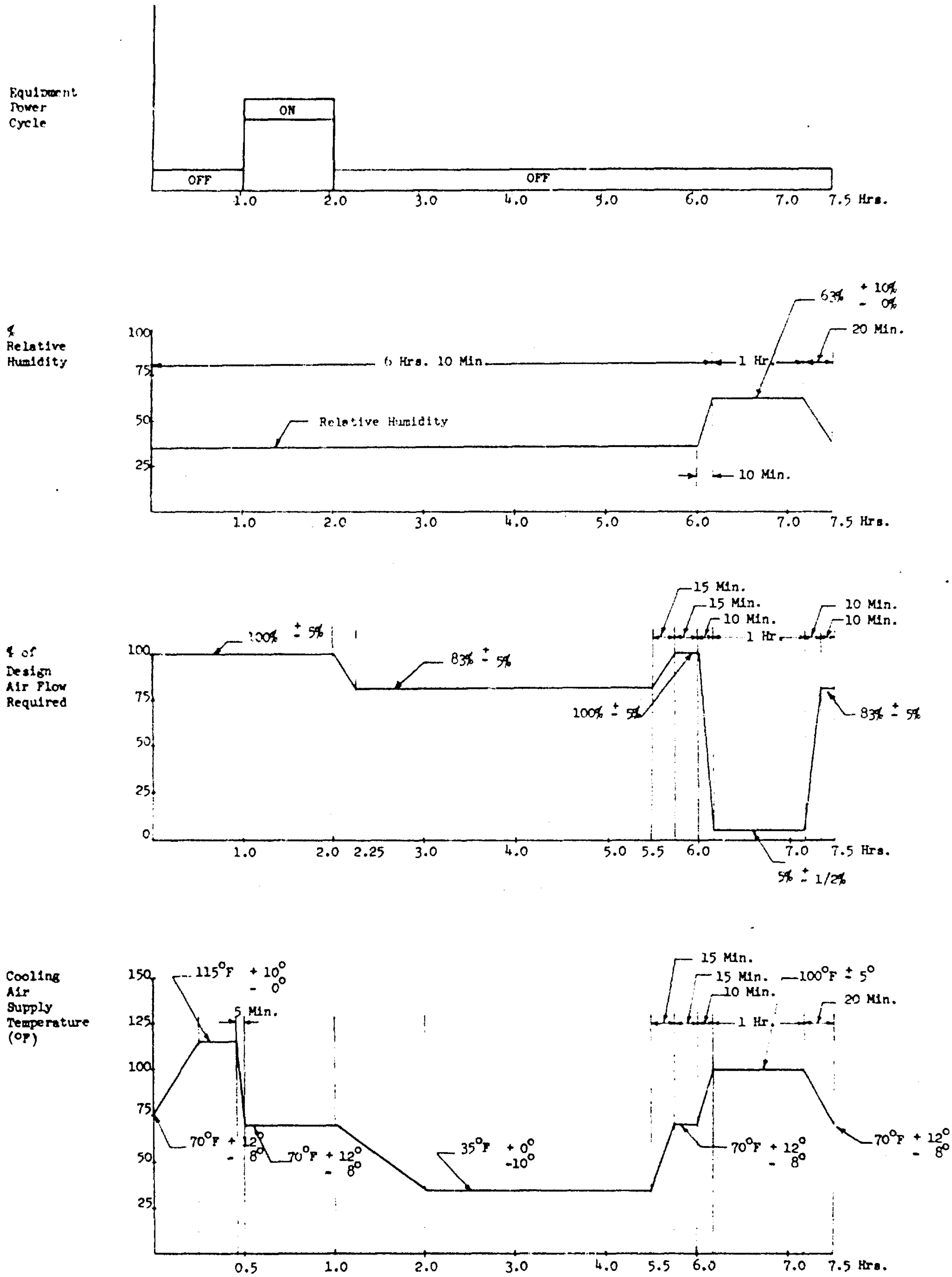


FIGURE 4 TEST CYCLE A - FORCED AIR COOLED EQUIPMENT

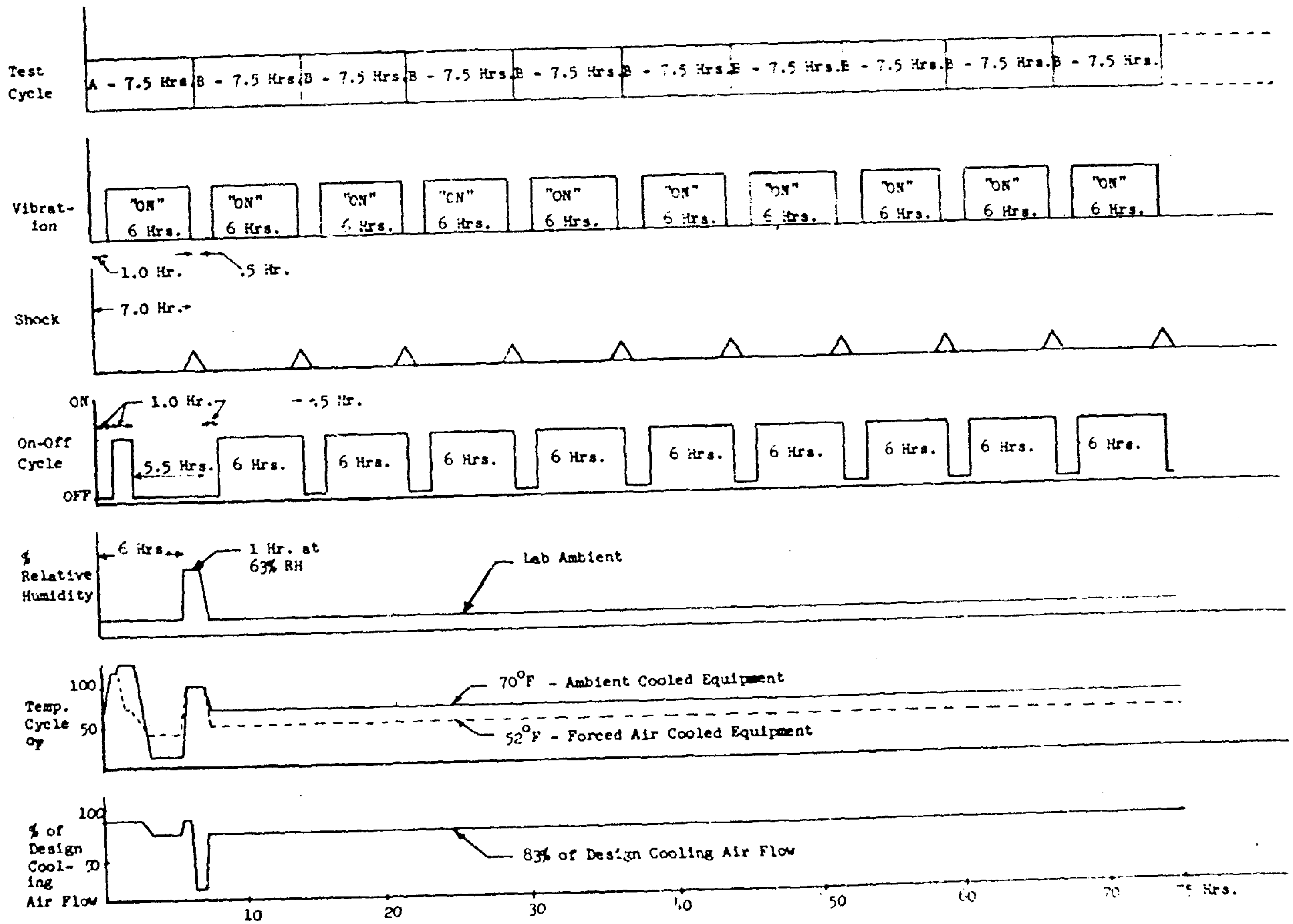


FIGURE 5 COMPOSITE ENVIRONMENTAL TEST CYCLE

Shock was applied at the end of each cycle at levels of 4.0 g vertical and 2.8 g lateral. The vibration environment was maintained for six hours for each "A" and "B" cycle and consisted of sweeps 10-500-10 Hz. and dwells at 73.6 and 147.2 Hz.

Levels were:

10 -41 Hz. at 0.018" D. A.

41 -500 Hz. at $\pm$ 1.5 g

3.4 Field Environments

Current aircraft weapons systems are exposed to a variety of natural and induced environments in the field. The magnitude, duration and recurrence of these stresses are a function of many operational parameters, some of which are predictable and others which are not. Ideally, new equipment designs incorporate reasonable margins of safety to permit successful operation during or after such exposures. In actuality, however, some of these design practices are not completely adhered to because of other, equally critical, programmatic constraints (e.g., cost, weight, power, location, etc.). Equipment may be subjected to conditions which are neither expected nor capable of being protected against and consequently not tested for. When equipment is then deployed in the field, a certain percentage of them failed because of the local conditions. These environmentally induced failures may be attributable to the stress level or duration or because of the coincidence of the stress with other operational conditions. In order to correlate the field data to particular non-benign conditions, potential environmentally induced equipment failure mechanisms were identified and are summarized in Table 4. The study performed by Grumman for the Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base (Reference (1)), plus a review of current laboratory and field data, indicate that of the total possible environments existing, seven are responsible for essentially all of the failures related to environmental causes. Figure 6 depicts the distribution of failures as a function of these environments. Temperature, vibration and moisture (including salt spray) accounted for 90% of the environmental failures. Thus, the study effort concentrated on the investigation of these environments and, to a lesser degree, evaluated the

TABLE 4 - POTENTIALLY DEGRADING ENVIRONMENTS AND PROBABLE EFFECTS ON AVIONIC EQUIPMENT

ENVIRONMENT	CONDITION (TYPE)	HOW MANIFESTED	PRINCIPAL EFFECT	PROBABLE FAILURE MODE
Temperature	Steady State - High	Ambient exposure Equipment induced Mission induced (certain steady state phases)	Aging Insulation deterioration Oxidation Expansion Reduction of viscosity Softening Evaporation, drying Chemical changes	Alteration of properties Shorting Rust Physical damage, increased wear Loss of lubrication in bearings, seizing Physical breakdown Dielectric Loss
	Steady State-Low	Ambient exposure Mission induced for certain phases and certain equipment Note: This condition is minimal. Encountered at present active bases, and in some flight modes	Contraction Viscosity increase, embrittlement, ice formation	Wear, structural failure, binding Loss of lubricity Structural failure, cracked components Structural failure, alteration of electrical properties Loss of resilience - seal leaks
	Thermal Cycling	Ground operation Mission operation and profile environment control system limits	Decrease in component reliability	Repeated stress variation causes mechanical failure of components solder joints, lifting of I/C's from base material
	Thermal Shock	Mission profile Geography and season of year	High temperature gradients	Mechanical failure • Cracks • Rupture

TABLE 4 - POTENTIALLY DEGRADING ENVIRONMENTS AND PROBABLE EFFECTS ON AVIONIC EQUIPMENT (Continued)

ENVIRONMENT	CONDITION (TYPE)	HOW MANIFESTED	PRINCIPAL EFFECT	PROBABLE FAILURE MODE
Vibration	Sine	Engine induced - Propeller aircraft	Force variation Periodic variation (motion is harmonic) • Mechanical stress • Fatigue	Structural failure Increased wear Interference with proper operation Relay, switch contact chatter
	Random	Engine Induced-Jet Aircraft Acoustic Noise Turbulence (Aeronautical Buffeting) Gunfire	Force variation - Random variation of amplitude and frequency Pressure loads and force • Mechanical stress • Fatigue	Same as Sine
Contamination	Sand and Dust	Sand lifted by wind. Dust particles present above desert areas and in atmosphere throughout world. Note: Confined to 10K' (maximum conditions at 1500')	Abrasion Clogging Sticking	Erosion of surfaces. Increased wear (especially in combination with moisture - water, oils, greases). Functional interference, arcing of high-voltage electrodes.
	Atmospheric Pollution	Chimney smoke Milling operations Volcanic action	Same as Sand and Dust, except formation of acids-in combination with mist.	Same as Sand and Dust plus extensive effects of acids
Explosive Atmosphere	Combustibles	Presence of combustible (fuel) gases inside equipment at temperature, R.H., & atmospheric pressure which favor explosion	Structural, etc. damage and/or complete destruction	Function interference. Loss of aircraft
Shock		Arrested Landing - Catapult Launch	Same as vibration	Same as vibration

TABLE 4 - POTENTIALLY DEGRADING ENVIRONMENTS AND PROBABLE EFFECTS ON AVIONIC EQUIPMENT (Continued)

ENVIRONMENT	CONDITION (TYPE)	HOW MANIFESTED	PRINCIPAL EFFECT	PROBABLE FAILURE MODE
Accelera-tion	Steady State	Catapult Launch - Maneuvers Due Mission Profile	Mechanical stress Induced switching, etc.	Loss of mechanical strength Interference of (relays, switchings, centrifugal devices)
Fungus		High R. H., optimum temperature (100°) plus nutrient material. (Tropical Environment)	Attack on organic materials	Loss of dielectric strength Electrical degradation
Bench Han-dling	Shock	Handling during shipping, installation, repairs, etc.	Structural damage Mechanical stress	Component damage, functional interference Electrical degradation (shorts, misalignment).
Atmospheric Electricity	Static	Autogenous-Rubbing of particles (snow, dust, sand) against vehicle surface. Exogenous-High potential gradients in atmosphere.	Personnel shock Combustibles- ignition Arcing Radio interference	Interference with duty Explosion (See explosive atmosphere) Shorting - component damage primarily semiconductors Mission interference
	Lightning	Difference in electrical potential between ground and clouds or cloud to cloud (within thunder clouds)	Surface damage; complete desctruction of non-metallic parts. Electrical damage	Loss of control surfaces Explosion of radomes, wind-shields. Current fed to electronic equipment from antenna causes equipment and component damage.
Radiation	Solar (Sunshine)	Heat energy leaving sun	See Temperature	See Temperature
	Cosmic	Sun, other sources. These are rays with enough energy to reach earth	Short term ionization. Generally no serious effects.	Spurious electrical pulses which may affect computers.

TABLE 4 - POTENTIALLY DEGRADING ENVIRONMENTS AND PROBABLE EFFECTS ON AVIONIC EQUIPMENT (Continued)

ENVIRONMENT	CONDITION (TYPE)	HOW MANIFESTED	PRINCIPAL EFFECT	PROBABLE FAILURE MODE
	Nuclear	Nuclear engines, nuclear reactors, nuclear weapons	Since frequency of occurrence is almost nil, this environment will not be considered.	
TRANSPORTATION	Land Truck Rail Air Conditioned compartment Non-conditioned compartment Sea	Shipment of equipment Delivery of equipment Shipment of equipment.	In general the effects can be established from other basic environments i.e., temperature, altitude, vibration, shock, etc. However, for air transport at 50K', a temperature of-105°F can be experienced per AT 70-38 (Army regulation)	See individual environmental Temperature Altitude Vibration Shock Susceptible equipment can be physically damaged even in the non-operating-storage state.
Moisture	Humidity	Water content of air	Galvanic action Microbiological growth Electrolysis Moisture absorption Corrosion	Loss of electrical properties Interference with function, swelling, rupture Dissolution of metals Increased wear Fungus growth and material damage
	Condensation	Variation in altitude causes condensation on structure and within equipment.	Same as Humidity	Same as Humidity
	Rain	Precipitation of water vapor	Same as Humidity, plus physical stress erosion	Same as Humidity, plus physical damage erosion

TABLE 4 - POTENTIALLY DEGRADING ENVIRONMENTS AND PROBABLE EFFECTS ON AVIONIC EQUIPMENT (Continued)

ENVIRONMENT	CONDITION (TYPE)	HOW MANIFESTED	PRINCIPAL EFFECT	PROBABLE FAILURE MODE
	Icing	Liquid droplets present at sub-freezing temperatures. Supercooled clouds.	Physical stress Added weight changes in aerodynamic profile	Structural failure Physical and electrical property changes Loss of performance or even entire aircraft. Interference with certain functions.
	Hail	Developed in thunderstorms	Physical damage	Dents, cracks, ingestion
	Salt Fog Salt Spray	Salt in suspension in water droplets, coastal areas, ocean atmosphere. Shipboard environments, high winds, etc., creating salt water spray.	Corrosion Electrolysis	Increased wear Dielectric loss Structural defects • Surface deterioration Increased conductivity
Pressure	Ambient	Altitude Variation	Outgassing force due to pressure differential. Reduced dielectric strength of air	Loss of lubricants Structural damage Corona discharge causes ozone formation and damages parts, causes insulation breakdown. Ozone oxidizes, rubber and synthetics also forms corrosive acids
	Explosive Decompression	Instantaneous loss of cabin compartment pressure	Large instantaneous pressure difference.	Mechanical damage and pressure shock to equipment.
	Wind	Differences in atmospheric density, producing horizontal difference in air pressure	Causes other environment to become dangerous. Sandstorms, blizzards alters flight paths of vehicles.	Primarily affects aircraft level. On ground could affect ground support equipment.

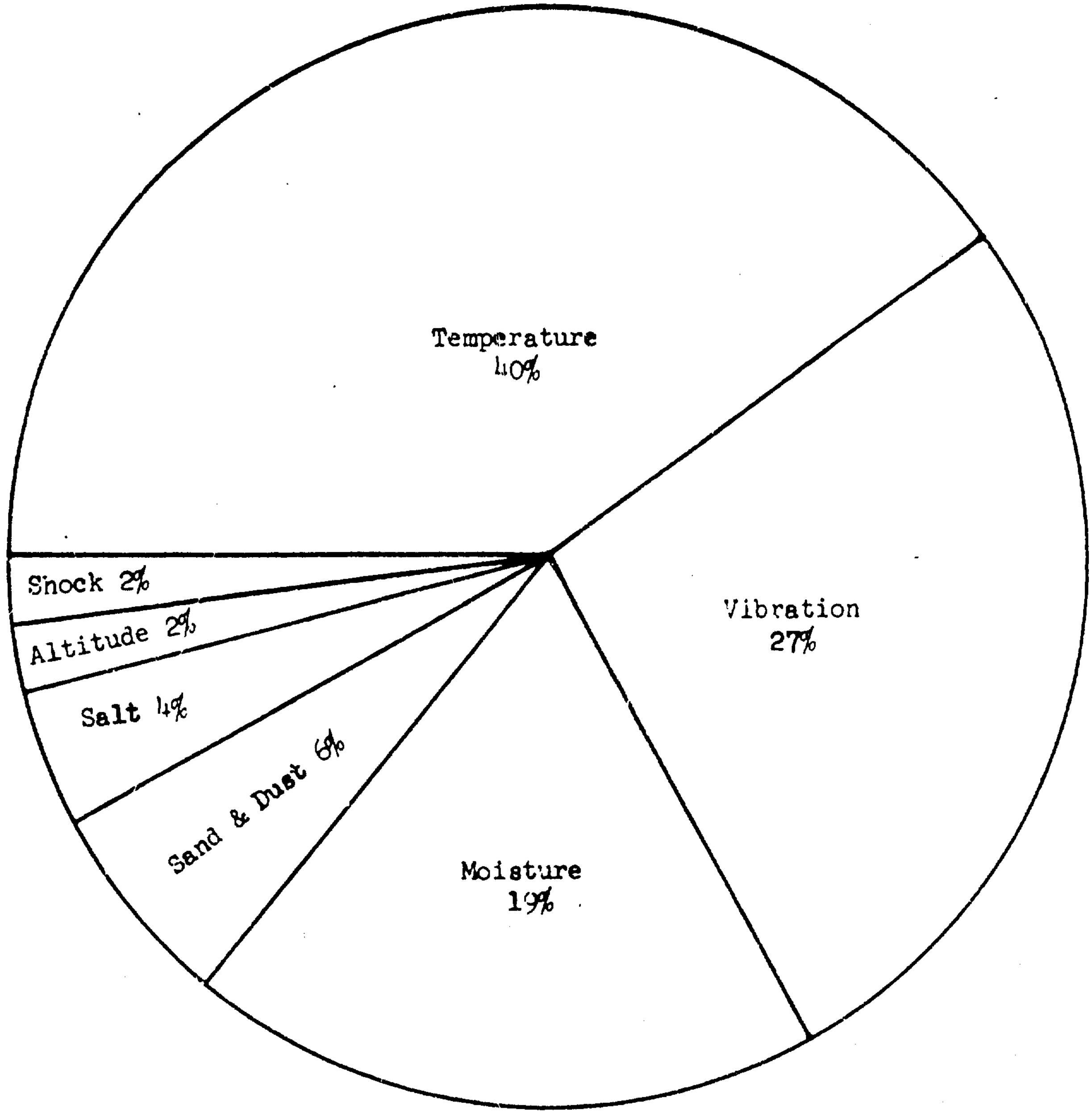


FIGURE 6 DISTRIBUTION OF ENVIRONMENTALLY RELATED FIELD FAILURES

effects of altitude and input power.

3.4.1 Thermal Environment

Physical Considerations - The parameters that define the thermal environment for a particular WRA are a function of the interrelationships that exist between the thermal characteristics of the WRA, the cooling methods employed and the features of the environmental control system. Thus, in order to assess the WRA thermal environment, some initial understanding of how these are considered in the design process is necessary. It is usually a design requirement to maintain aircraft compartment temperatures within the limits specified in MIL-E-5400. Parameters which affect the compartment temperature include the structural heat load due to aerodynamic heating (or cooling), the heat dissipated by the avionics and the flow rate and temperature of cooling air which enters the compartment. The cooling air may be provided by a combination of the aircraft air-conditioning pack, a ram air scoop and cascade flow from adjacent compartments. For compartments where dissipations are low, the temperature can be maintained simply by the aerodynamic heat leak. For this condition, the electronic heat load will exactly balance the aerodynamic heat load. Once the temperature limits have been specified for a given compartment, cooling requirements for the individual avionics boxes or groups of equipment must be investigated. The major consideration in the thermal design of avionics is the operating temperature limit of the electronic components within the box. This component temperature limit dictates the cooling method required, i.e., ambient cooling, fan cooled, forced air cooled or ram air cooled.

Ambient cooling refers to a box whose power dissipation is sufficiently low such that natural convection and radiation maintain all components below their operating limit.

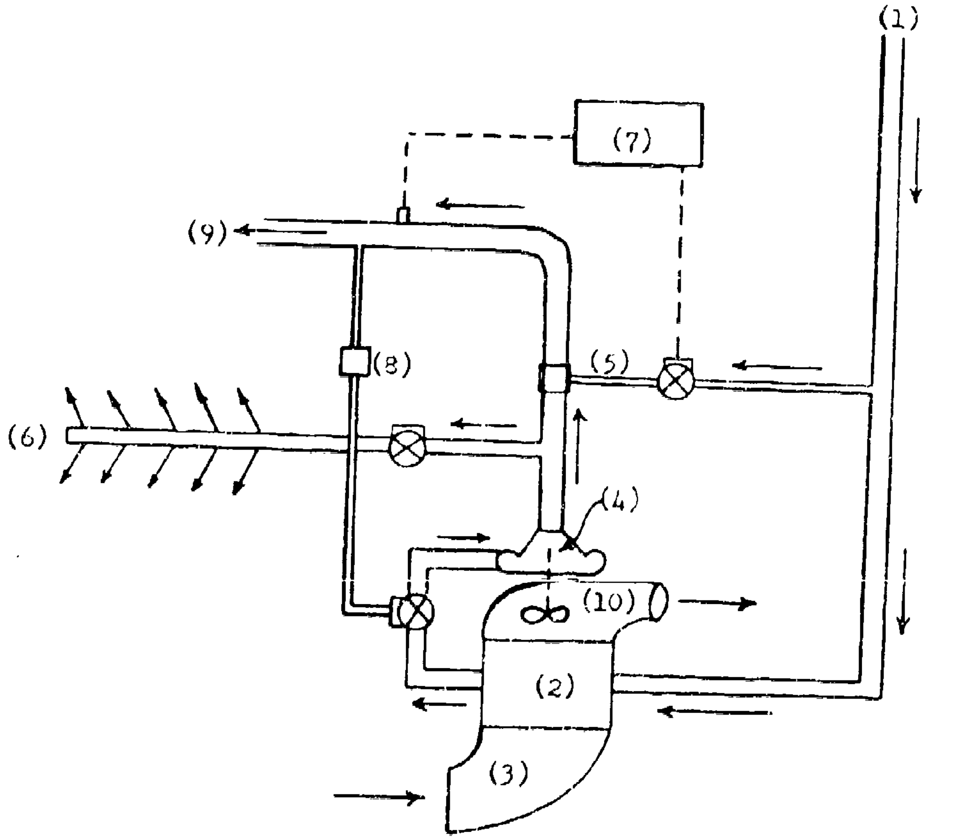
A fan cooled box is typically one whose power dissipation warrants circulation of the compartment ambient air through the unit to maintain it at satisfactory temperatures. The circulation, which is achieved by use of a blower, increases the heat transfer between the components and the ambient air.

Avionics boxes which have high power dissipation or which require accurate temperature control (such as gyros) are usually forced air cooled. This means that air at a controlled temperature and flow is forced through the box for control of its internal temperature. The cooling air can be supplied by the air-conditioning pack, a ram air scoop or cascaded flow from a compartment which is actively controlled (such as the cockpit). The thermal design of the box includes a specific requirement of cooling air flow rate variation.

Thus the definition of environment that an avionic equipment sees is dependent upon the cooling method. For ambient cooled boxes and boxes that contain an internal blower, the environment is totally defined by the compartment ambient temperature. The environment for forced air cooled equipments is defined primarily by the cooling air temperature and flow rate and secondarily by the compartment ambient temperature. For the particular case of equipments which are ram air cooled, the environment is defined by the ram air temperature and flow rate.

Since the local environment of a particular WRA is so dependent on the conditioned air, a brief understanding of the function and design of an aircraft environmental control system is presented to provide some insight into the thermal control process. The heart of an aircraft ECS is the air-conditioning pack, shown schematically in Figure 7, which provides the source of cooling air at a controlled temperature.

High temperature, high pressure air is bled from the aircraft power plant and passed through a ram air/bleed air heat exchanger where the temperature is cooled by the ram air. The cooled bleed air is then allowed to expand through a turbine where the air temperature is lowered by the expansion process. In the expansion process, the bleed air is allowed to do work on the turbine, thus resulting in lower temperatures. It is not uncommon for temperatures as low as $-65^{\circ}F$ to be achieved. The turbine outlet temperature can then be controlled by various means, such as addition of hot bleed air for reheating. Compartment temperature control and avionics cooling is provided by use of conditioned air used in conjunction with other thermal control devices and techniques, such as heaters, flow control devices and supplementary ram air cooling. Figure 8 illustrates some of the interrelationships that must be considered in the



LEGEND

(1) Engine Bleed Air
(2) Ram Air/Bleed Air Heat Exchanger
(3) Ram Air Intake Scoop
(4) Turbine
(5) Bleed Air Mix Line for Re-Heating Turbine Outlet Air
(6) Conditioned Air for Ram Air Cooled Compartment when Ram Air Temperature is High
(7) Temperature Controller/Sensor
(8) Flow Controller
(9) Conditioned Air for Avionics Cooling
(10) Ram Air Exhaust

FIGURE 7 SCHEMATIC OF A TYPICAL AIR CONDITIONING PACK

design of an environmental control system.

Increases in Ram Air Flow ⟹	Decreases Air Conditioner Turbine Outlet Temperature ⟹	Decreased Compartment Temperature
Increases in Ram Air Temperature ⟹	Increases Air Conditioner Turbine Outlet Temperature ⟹	Increases Compartment Temperature
	Increase in Air Conditioner Flow Rate ⟹	Decreases Compartment Temperature

FIGURE 8 ENVIRONMENTAL CONTROL SYSTEM PERFORMANCE MATRIX

As indicated previously, aircraft avionics are designed to perform within the temperature limits as specified by MIL-E-5400. While these limits may be approached under certain conditions, they are extremes in the flight envelope and in actual operation, compartment temperatures will be well within the high and low limits. In most instances, actual compartment temperature limits may be well below the operating limit on hot day and above the cold day limit. Hot day and cold day refer to the design levels of outside temperature versus altitude as specified in ANA Bulletin 421. The actual operating point of a compartment is established based on the combined effect of such operational parameters as altitude, Mach number and outside ambient temperature, as well as air conditioner performance characterists.

Table 5 summarizes the first order effects of the operational parameters on compartment temperature, ram temperatures and air conditioner performance. Each of the relationships shown assumes that all parameters remain constant except for the one being considered. While the Table generally tries to identify the major environmental effect of changes in aircraft altitude, aircraft speed and outside air temperature, many competing interrelationships exist. Thus, the net effect of variation in all parameters can only be arrived at by flight test measurement of detailed analysis. For example, the effect of

TABLE 5 EFFECT OF OPERATIONAL PARAMETERS ON ENVIRONMENTAL CONTROL PERFORMANCE

OPERATIONAL PARAMETERS	ENVIRONMENTAL CONTROL PARAMETERS				
	COMPARTMENT AMBIENT TEMPERATURE	AIR CONDITIONER TURBINE OUTLET TEMPERATURE	AIR CONDITIONER AIR FLOW RATE	RAM AIR FLOW RATE	RAM AIR TEMPERATURE
INCREASING ALTITUDE	DECREASES	DECREASES	DECREASES	DECREASES	DECREASES
INCREASING AIR SPEED	INCREASES	CANNOT BE GENERALIZED	INCREASES	INCREASES	INCREASES
INCREASING STATIC AIR TEMPERATURE (SEA LEVEL)	INCREASES	INCREASES	DECREASES	DECREASES	INCREASES

increasing aircraft speed on air-conditioner turbine outlet temperature can not be generalized until all the system characteristics are defined (e.g., the specific design of the air conditioner ram air heat exchanger). An increase in the aircraft's velocity will result in an increase in ram air temperature as well as an increase in ram air flow rate. The resulting temperature of engine bleed air exiting the heat exchanger can be higher due to the increased air temperature from the engines and increased ram air temperature, or it can be lower due to the increase in the ram air flow rate. As a result, one cannot generalize in advance about the effect of velocity increase on air-conditioner turbine outlet.

<u>Data Collection and Analysis</u>

Compartment ambient temperatures are measured during the flight test program for various portions of the flight profile. Based on these measured temperatures, heat transfer coefficients are calculated to provide heat infiltration or loss to adjacent compartments and infiltration or loss through the skin of the aircraft. Using these calculated values of heat transfer coefficients, the observed compartment ambient temperature can then be extrapolated to hot and cold day conditions. These latter values are those which have been used in this study.

The following sample calculations indicate how the ambient temperature extremes were determined for the nose compartment of one of the study aircraft. The equation for overall heat balance is:

$$Q_{AERO} + Q_{ELECT.} + Q_{FLOW\ IN} - Q_{FLOW\ OUT} = 0$$

where: Q_{AERO} = Aerodynamic Heating (or Cooling)

$Q_{ELECT.}$ = Power Dissipation

$Q_{FLOW} = Q_{FLOW\ IN} - Q_{FLOW\ OUT}$ = Thermal Flow Energy Into or Out of Compartment

where: $= \dot{m}\ Cp\ (T_{IN} - T_{COMPT})$

$\dot{m}$ = Mass Flow Rate

C_p = Specific Heat

T_{IN} = Flow Temperature In

$T_{COMPT.}$ = Compartment Temperature

Substituting for Q_{FLOW}:

$$Q_{AERO} + Q_{ELECT.} + \dot{m}\ C_p\ (T_{IN} - T_{COMPT.}) = 0$$

Values for each of the equation terms were obtained as follows:

- $Q_{ELECT.}$ is the measured (laboratory) power dissipation of the avionic equipment located in the compartment.
- T_{IN} and $\dot{m}$ were obtained from laboratory tests of the actual flight cooling system.
- Q_{AERO} was obtained from flight data heat load curves which are a function of ambient temperature, Mach number, altitude and compartment ambient temperature. For the maximum compartment temperature (sea level and V_{MAX} conditions), the heat load curves for extreme hot day conditions (Figure 9) were used. For minimum compartment temperature (altitude and minimum velocity to preclude aircraft stall) Q_{AERO} for an extremely cold day was determined from the heat load curves of Figure 10.

Utilizing the values so obtained, $T_{COMPT.}$ was then calculated from the heat balance equation. The final determination was an iterative process wherein a compartment temperature was assumed and the heat balance equation solved. This process continued until the assumed value equaled the calculated value. To illustrate how this procedure was followed for maximum temperature conditions, assume

Q_{ELECT} = 5620 BTU/Hr.

$\dot{m}$ = 2280 lbs./hr. at 80°F

C_p = 0.25 BTU/lb. - $^{\circ}$F

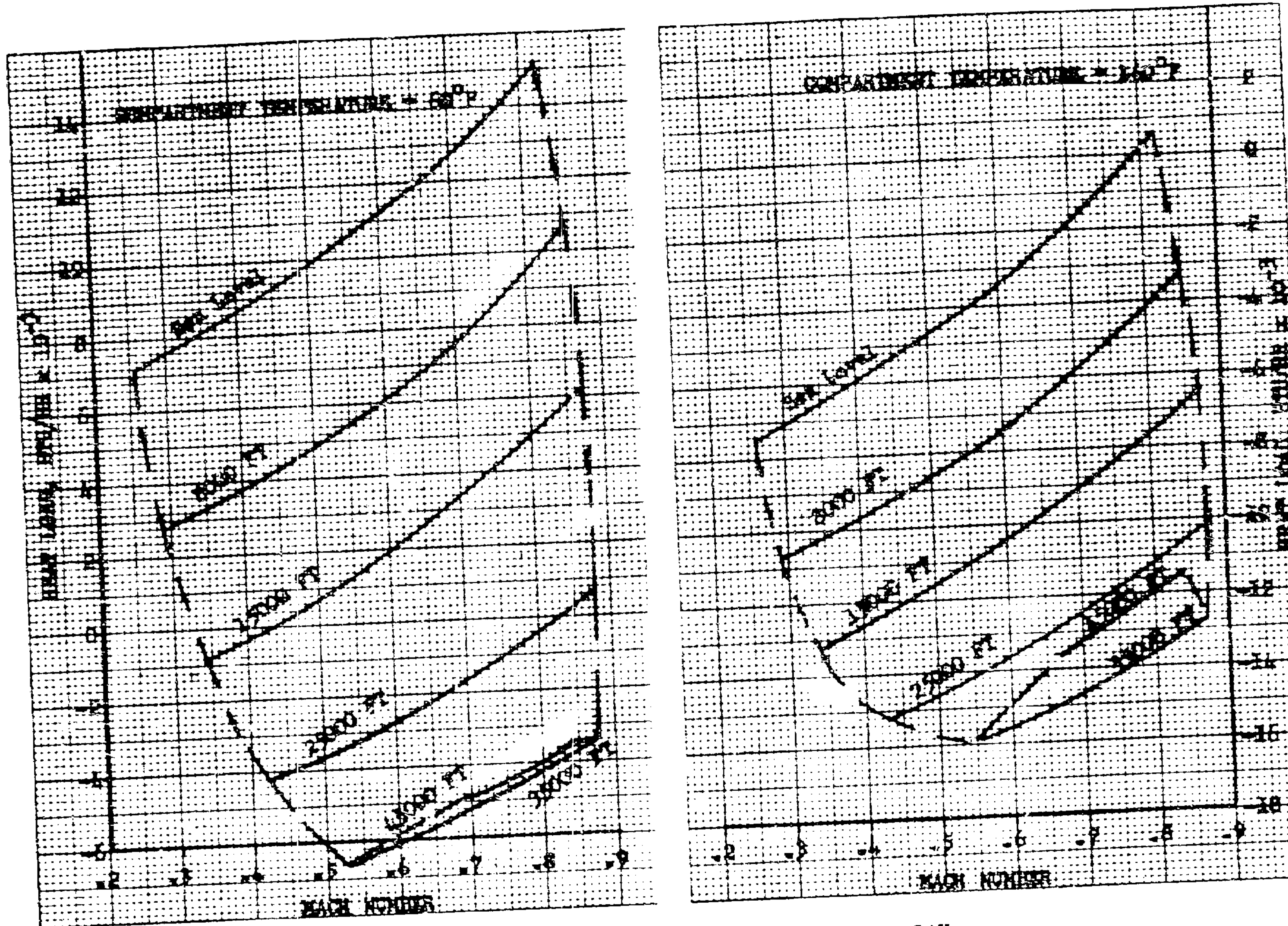


FIGURE 9 TYPICAL AIRCRAFT HEAT LOAD -- HOT DAY

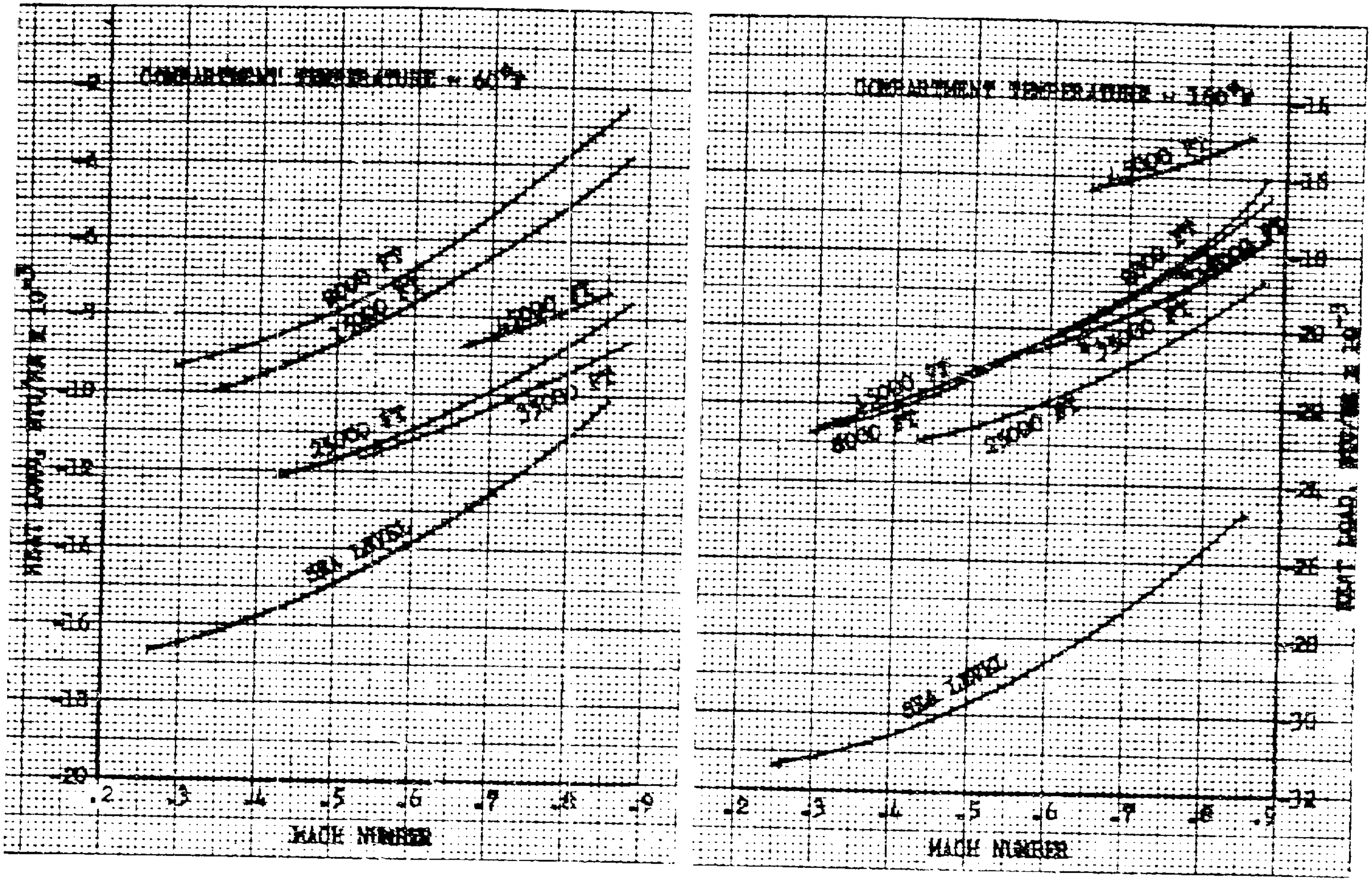


FIGURE 1C   TYPICAL AIRCRAFT HEAT LOAD -- COLD DAY

From Figure 9: Q_{AERO} = 15,200 BTU/hr. at a compartment temperature of 60°F.

Q_{AERO} = 600 BTU/Hr. at a compartment temperature of 160°F.

Step 1

Assume: $T_{COMPT} = 120^{\circ}$

Q_{AERO} at this temperature is determined by interpolation between the given values.

$Q_{AERO} = 6440$

Solving the heat balance equation:

$$0 = 6440 + 5620 + (2280)\ (.25)\ (80 - T_{COMP})$$

From which:

$T_{COMP} = 101^{\circ}$

But: Computed value (101°) $\neq$ assumed value (120°)

Step 2

Assume: $T_{COMPT} = 100^{\circ}$

Then: Q_{AERO} (by interpolation) = 9360

Solving:

$$0 = 9360 + 5620 + (2280)\ (.25)\ (80 - T_{COMP})$$

From which:

$T_{COMPT} = 106^{\circ}$

But: $106^{\circ} \neq 100^{\circ}$

Step 3

Assume: $T_{COMPT} = 105^{\circ}$

Then: Q_{AERO} (by interpolation) = 8630

Solving:

$$0 + 8630 + 5620 + (2280)(.25)\ (80 - T_{COMPT})$$

From which:

$$T_{COMPT} = 105^{o}$$

Since the computed value equals the assumed value the process stops and $105^{o}F$ is the maximum compartment temperature.

The above type of analysis applies to compartments which have no active means of temperature control. For a controlled compartment (such as the cockpit), the design limits are given and have been verified during the flight test program.

As previously indicated, the thermal environment experienced by a black box is a function of mission profile and location within the aircraft. One common phenomenon which has been observed is that all compartment ambient temperatures, experience, to some extent, a thermal cycling effect whose frequency is much greater than that required by MIL-STD-781. To illustrate this, two typical curves are presented for compartment temperature variations as a function of altitude and speed. Figure 11 shows thermal variations within a fuselage compartment of a turbojet aircraft. It can be seen that there is a corresponding change in the absolute value of compartment temperature when steady state values of speed and altitude changes. In addition, when rapid variations in these flight parameters occur, thermal effects become apparent.

Figure 12 represents the nose compartment thermal profile for another turbojet aircraft and indicates a wide variation in temperature for changing flight conditions. In this case the two cyclical variations of approximately 30 to 40 minutes duration are apparent. By comparison, the thermal cycle presently defined by MIL-STD-781 encompasses a six hour period.

Forced air cooled boxes effectively shield the electronic components from changes in compartment ambient temperature. The operating temperatures of the components within the box are usually established by the temperature and flow rate of the cooling air.

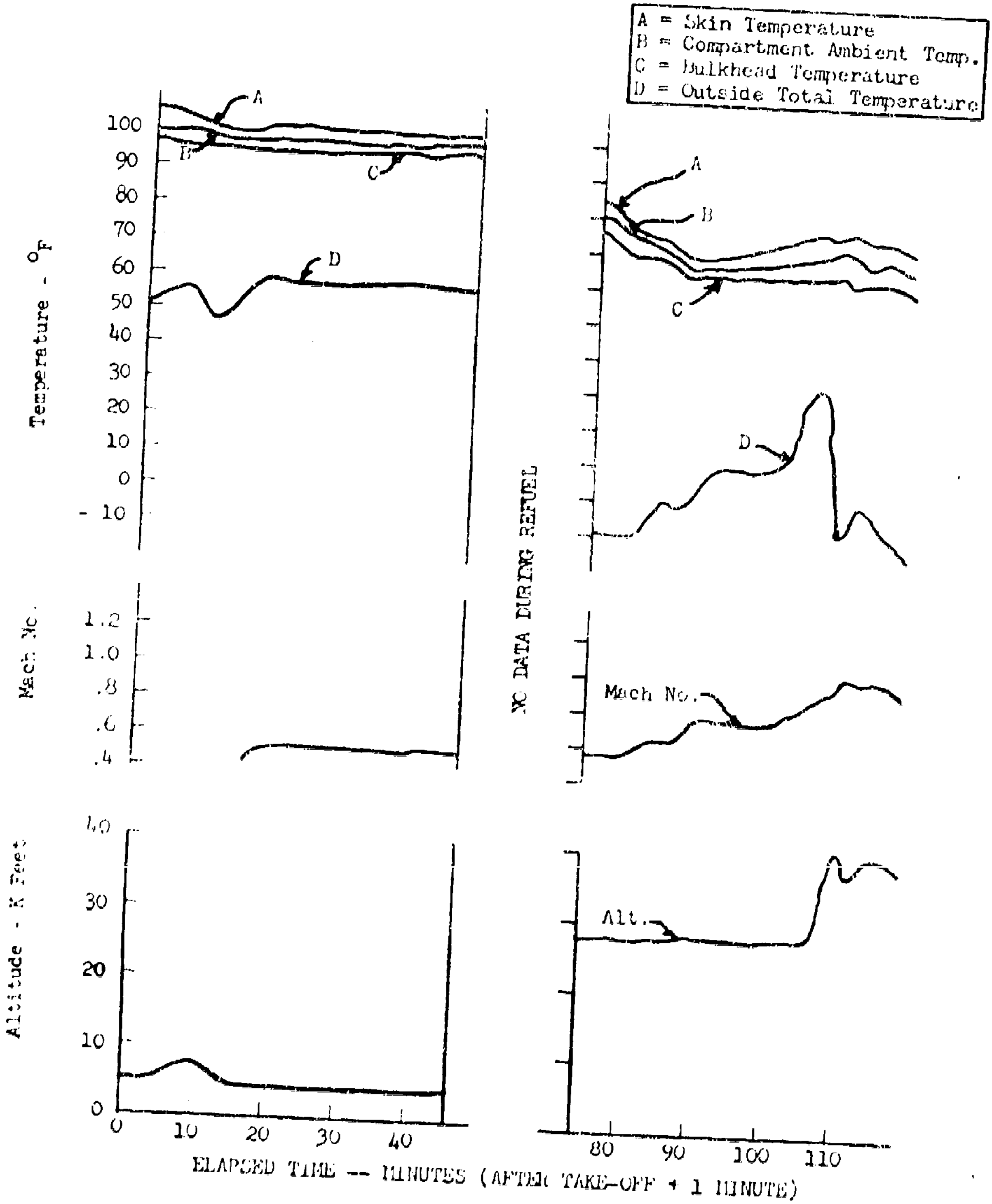


FIGURE 11 TYPICAL AIRCRAFT TEMPERATURE PROFILES

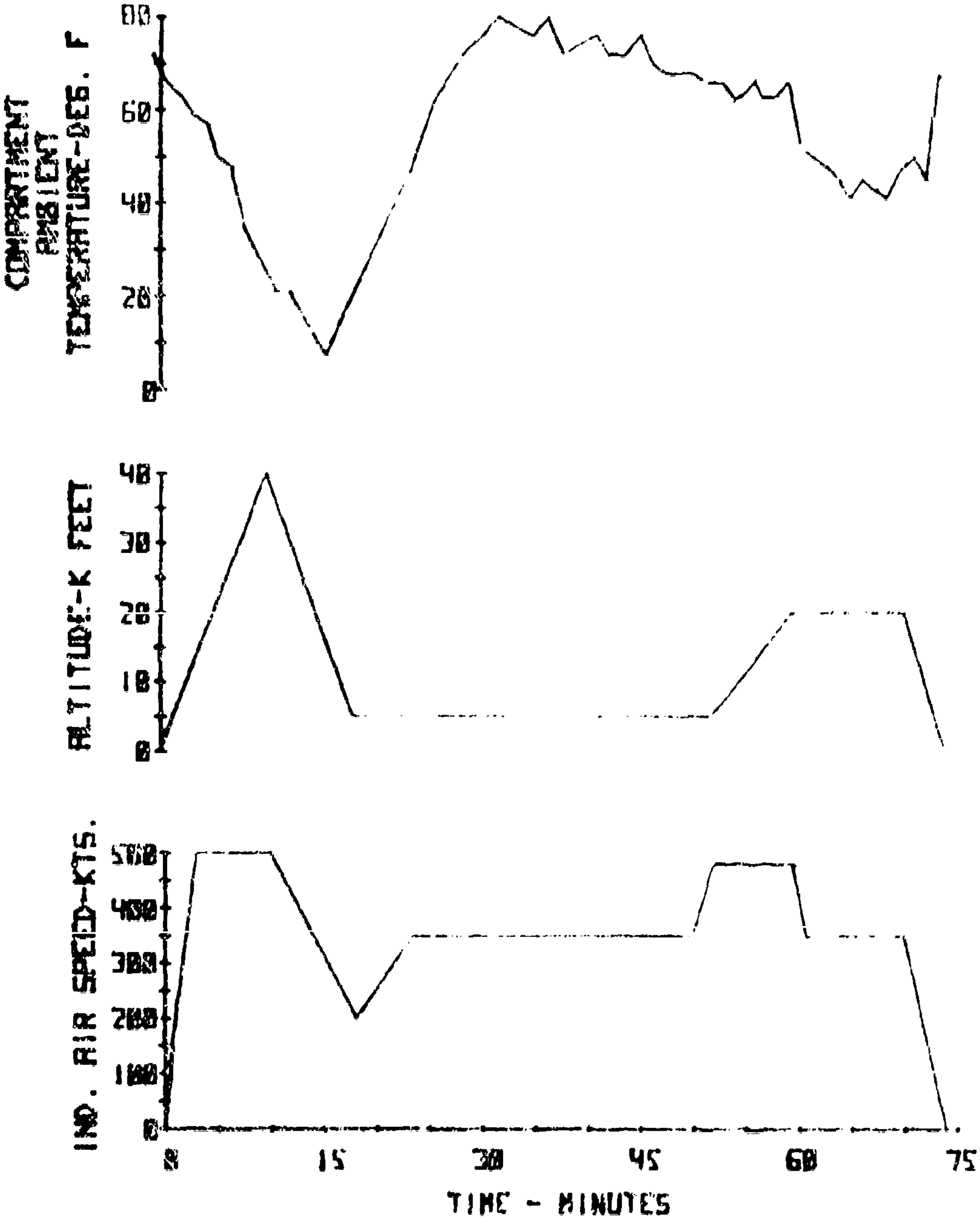


FIGURE 12 TYPICAL NOSE COMPARTMENT TEMPERATURE VS. MISSION PROFILE

The cooling air flow rates are based on measured data taken during laboratory tests and later verified in the flight test program. Separate tests were conducted to determine the performance of the air conditioning pack and flow splits to the various avionics boxes. The flow required for each fan cooled box is a function of the power dissipation and the cooling air supply temperature. The worst case condition for each box is determined and the required flow to the box is supplied. At all other conditions, the box will receive a flow in excess of that required to satisfy the heat load and therefore will cool the box.

The flows established in the laboratory testing are then verified in actual flight testing. A typical plot of cooling air temperature is provided in Figure 13. The data represents two minutes of dynamic maneuvering during a test flight of a turbojet aircraft and shows that, although the cooling air supply temperature is controlled within specified limits, some thermal transients are evident. In utilizing this data, it must be understood that it represents absolute temperatures and temperature variations observed within the control loop of the air conditioner system. The data was obtained by sensors in the outlet airstream and therefore does not include the damping effects, imposed by the downstream ducting and other thermal masses, which may be evidenced at the inlet of a particular WRA. This representation of the typical response of the air conditioner control loop indicates that although sharp instantaneous rates of change (up to 90°F/minute) do exist undamped for several seconds, at no time during these transients does the absolute cooling air temperature vary outside of the control setting of 35 ± 5°F. This type of variation is characteristic of environmental control systems.

Data Presentation

Tables B-1 and B-2 of Appendix B present the laboratory and field thermal environments for each black box. Table B-1 includes the compartment ambient air temperature parameters and Table B-2 presents the pertinent cooling air and thermal parameters for those WRA's requiring supplemental cooling. These values were obtained in the manner described above and represent the extreme conditions based on cold day/coldest mission and hot day/hottest mission. While the rates of change presented in the Tables cover

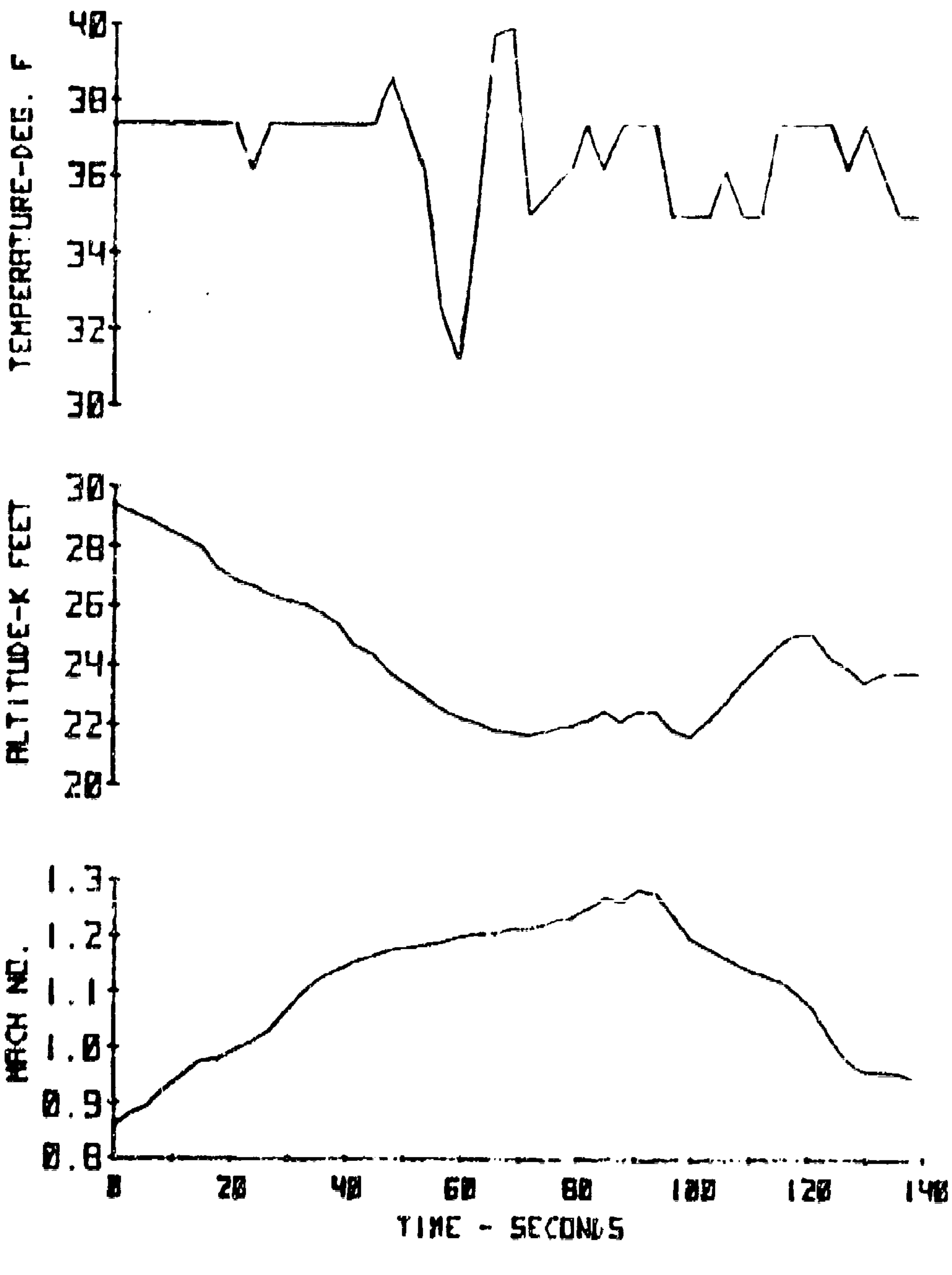


FIGURE 13 COOLING AIR TEMPERATURE VS. MISSION PROFILE

the average rates during a maneuver, instantaneous rates of change can be much higher for very short durations. Therefore, since the data presented in the Tables represent those steady state conditions which occur at extremes in flight envelope, the average rates presented should be used in the transition from one point to the other. The indicated duration at temperature extremes were referenced to laboratory test time and were calculated by defining the percentage of time the aircraft remained at similar extremes (for a composite mission) and multiplying this percentage by total test operating hours. The values so obtained could then be compared directly with laboratory test durations under similar conditions. Rates of change listed for each item are the maximums encountered in flight. The delta columns represent the difference between laboratory and field and the rule established was "field minus laboratory." Hence, a positive sign denotes that the field value was greater and a negative sign indicates a lower field condition. For temperature differences a negative sign was bracketed to avoid confusion with sub-zero temperatures.

Demonstration/Field Comparisons

A summary of compartment ambient air parameter comparisons between the laboratory and the field is shown in Table 6. It indicates that the bulk of the ambient cooled WRA's see a field temperature environment that is less extreme and for a shorter duration than they are exposed to in the laboratory. Most of the group, however, experienced greater rates of change in the field. The same conclusions were reached for forced air cooled WRA's with the exception of the high temperature level and total duration. It appears that more of these WRA's experience higher ambient temperatures for longer periods of time in the field than in the laboratory. This, however, is to be expected since the operational consideration of anticipated high temperature (ambient) dictated the requirement for supplemental cooling.

A similar survey of cooling air parameters between laboratory and field for those WRA's that are supplementary cooled is presented in Table 7. It shows that the cooling air temperature and the rate of change thereof is at least as extreme as in the field, for the greater majority of the WRA's. Flow extremes appear to be higher in the field, in that more WRA's

TABLE 6 SUMMARY COMPARISON OF COMPARTMENT AMBIENT AIR PARAMETERS

COMPARTMENT AMBIENT AIR PARAMETER	QUANTITY OF WRA's					
	AMBIENT COOLED EQUIPMENT		SPECIALLY COOLED EQUIPMENT		TOTAL	
	LABORATORY LESS THAN FIELD	LABORATORY GREATER THAN FIELD	LABORATORY LESS THAN FIELD	LABORATORY GREATER THAN FIELD	LABORATORY LESS THAN FIELD	LABORATORY GREATER THAN FIELD
Minimum Temperature	45	7	21	22	66	29
Maximum Temperature	12	40	33	10	45	50
Duration of Low Temperature	8	44	26	17	34	61
Duration of High Temperature	14	38	31	12	45	50
Temperature Rate of Change	42	10	41	2	83	12

TABLE 7 SUMMARY COMPARISON OF COOLING AIR PARAMETERS

COOLING AIR PARAMETER	QUANTITY OF WRA's		
	LABORATORY LESS THAN FIELD	LABORATORY EQUAL TO FIELD	LABORATORY GREATER THAN FIELD
Maximum Temperature	3	25	15
Minimum Temperature	23	8	12
Temperature Rate of Change	2	17	24
Minimum Flow Rate	22	9	12
Maximum Flow Rate	25	9	9
Duration	11	22	10

had minimum and maximum flow rates greater than the corresponding laboratory rates. Higher flow rates at a given cooling air temperature tend to provide more cooling capacity. However, the higher flow rates do give rise to high thermal gradients and more rapid transients during environmental changes. As a result, it is possible that the higher flow rates are subjecting the internal components of the WRA to higher stresses in the field.

3.4.2 Vibration

In order to determine the dynamic environment experienced by a WRA, some understanding of the definitions, specific sources, and operational factors influencing this environment is first necessary.

The input energy to an equipment may generally be random, sinusoidal, or a combination of both in character. Random vibration is that displacement whose instantaneous magnitude is not specified for any given instant of time. The instantaneous magnitudes of a random vibration are specified only by probability distribution functions giving the probable fraction of the total time that the magnitude lies within a specified range. Random vibration contains no periodic or quasi-periodic components. Sinusoidal vibration, on the other hand, is a simple harmonic motion such that the displacement is a sinusoidal function of time. Since the combination of both is observed in aircraft, it is important to examine the time history for each data point prior to determining the method of data analysis. In the case where there are steady state events, specifically where any dynamic system is acted upon by a definite force system that will follow a definite cycle of events, such as those that occur during takeoff, cruise and high speed flight, a narrow-band Power Spectral Density (PSD) analysis is recommended. These PSD playbacks reveal a broadband random base with a series of sinusoidal responses at specific frequencies. For transient events, i.e., when any phenomena which occur during the time required for the response to adapt itself from one force system to another, such as those situations during catapult launch, arrested landing, buffet, abrupt manuevers, etc., a transient data analysis that examines the peak values and the number of occurrences is performed, and its damage potential is evaluated as a function of aircraft mission

or event exposure.

The vibration input a WRA experiences is the combined effect of several mechanical, acoustic, and aerodynamic phenomena. The aircraft's propulsion system is one of the major sources of vibration. It produces vibration both mechanically and by generating noise. Mechanical vibrational responses from the engine are due to the unbalance existing in the rotational parts of the engine. The fundamental frequencies and their harmonics are directly proportional to the operating speeds of the engine. Noise vibrations are caused by the turbulent mixing of the high velocity exhaust gas with the ambient atmosphere.

Auxiliary systems such as electric motors, hydraulic pumps and actuators, environmental control systems, etc., also cause vibration. The vibration responses are due to the unbalance in the rotating parts and the frequencies directly proportional to their operating speeds.

Buffet and boundary layer effects are generally the major contributors to vibration from aerodynamic influences. Buffet vibrations result when the structure of the aircraft is forced to respond when subjected to unsteady aerodynamic forces. Boundary layer vibrations are due to the pulsations in the turbulent boundary layer that result from dominant sound pressure loads impinging on the aircraft.

Additional sources of vibration resulting from specific mission or operational conditions of use are gunfire, taxiing, catapult launch, and arrested landing. Gunfire caused vibrations are due to the firing rate of the gun. The gunfiring forces contain integral harmonics of the fundamental firing rate frequency and the responses include the effects of impulsive loadings on the structure by gun-muzzle blast. Taxiing vibrations are caused by vertical response of the airplane to irregularities and waviness of the taxiway surfaces. During catapult launch, structural dynamic response and transient vibrations occur when a sudden auxiliary thrust is added to the aircraft's thrust to effect a launching into flight with minimal travel down a runway. Arrested landing produces dynamic transient vibration when the aircraft lands and its forward velocity is abruptly halted by the aircraft's arresting hook catching a deck cable.

Inasmuch as the propulsion system has been identified as one of the major sources of vibration, the aircraft selected for this study had different propulsion systems and flew different types of missions. Aircraft equipped with the following engines were selected for this study: turbojet; turbofan with afterburner; and constant speed turboprop. A turboprop engine is a gas turbine in which the turbine provides power in excess of that required to drive the compressor, and that excess power is used to drive a propeller constituting approximately 80% of the total thrust with the remaining 20% from jet thrust. A turbojet engine is a gas turbine in which no excess power (above that required by the compressor) is supplied by the turbine. The available energy in the exhaust gases is converted to kinetic energy of the jet. A turbofan is similar to a turboprop except that the excess power is used to drive a fan or low pressure compressor in an auxiliary duct, usually annular around the primary duct. In an aft turbofan, an aerodynamically coupled turbine is usually used to drive the low pressure ratio duct compressor (fan). An afterburner can be used in any of these above devices to give additional thrust at the expense of fuel economy. Additional fuel is added to the exhaust gases and burned, thereby increasing the temperature, the jet velocity, and the thrust. The turbojets and turbofan, herein referred to as jets, have engines mounted in the fuselage and the turboprop has one engine mounted on each wing.

Each of the aircraft chosen for this study was designed to accomplished a different primary mission. Thus the vibration levels measured in these aircraft is representative of a good cross section of aircraft environment. The primary mission and brief description of each aircraft is as follows:

- Aircraft No. 1 -- This aircraft is designed specifically for air defense. It has two turbojet engines without afterburners mounted in the mid-fuselage, and contains electronic equipment forward and aft of the engine compartment.

- Aircraft No. 2 -- This aircraft is designed specifically for attack missions. It has two turbojet engines without afterburners mounted in the mid-fuselage, carries various external missiles and has

a large number of electronic units mounted forward and aft of the engine.

- Aircraft No. 3 -- This aircraft is designed specifically for fighter missions. It has two turbofan engines with afterburners, mounted in the aft-fuselage, carries a wide variety of weapons, and has a large number of electronic units mounted forward of the engine.

- Aircraft No. 4 -- The primary mission of this aircraft is early-warning. It has a turboprop engine mounted on each wing and has electronic equipment installed throughout the entire fuselage.

Data Collection and Analysis

The information employed in this study was gathered to substantiate the operational environment for equipment installed in the fuselage of various aircraft. In particular, the vibration data examined in this study was only associated with the steady state portion of any given flight or mission. The conditions examined includes ground operations/take-off, cruise, high speed flight, and landing, but only those conditions producing the highest steady state levels were analyzed in detail. A review of the data analysis revealed the most severe conditions occurred at maximum engine power during ground operations/take-off and maximum speed at low altitude (5000 feet). Also included in the responses was the energy associated with the various on-board mechanical, electrical and hydraulic input sources (auxiliary systems).

Vibration measurements were acquired utilizing an Endevco piezo-electric crystal accelerometer that combines high sensitivity, broad temperature range, high resonant frequency and high capacitance into a lightweight reliable sensor. At each measurement location a tri-axial cluster of accelerometers was installed, such that the minimum resonant frequency of the installation was greater than 600 Hertz.

These instrumentation locations were chosen to describe the operational vibration environment for equipment installed on internal shelving in the study aircraft. The locations varied throughout the fuselage, from the nose to the rear of the aircraft. The data acquisition system for each aircraft consisted of a sixteen track hybrid tape recorder capable of

recording four tracks of proportional bandwidth analog data. The data was reduced on a (Ubiquitous) Real Time Power Spectral Density (PSD) System (up to 500 Hertz). This system is comprised of a highly sophisticated group of equipment for analyzing sinusoidal, transient, or random signals in the shortest time possible. Since the jet aircraft environment consisted of low level random and high level narrow band spikes, the choice of filter for PSD analysis was predicated on good data resolution. An accepted rule of thumb, developed from past experience, for choosing a filter bandwidth is to use one-quarter of the bandwidth of the lowest frequency response desired. Based upon that, a 3.3 Hertz bandwidth filter was utilized for the PSD analyses in this study. A data sample length of 20 seconds was analyzed yielding 128 degrees of freedom required for good statistical quality. In the case of the turboprop engine which rotates at a constant speed (1106 RPM), prior analyses indicate that the environment is predominantly sinusoidal. It was therefore possible to analyze 10-15 second time samples with 1.6 Hertz bandwidth filter to obtain good resolution.

Data Presentation

The equipment vibration data gathered for this study are presented in Appendix C, Figures 1 through 17, which are summarized below.

APPENDIX C FIGURE #	WEAPONS REPLACEABLE ASSEMBLY
1	54, 55, 56, 58, 60, 62, 67, 70, 72, 74, 77, 83
2	54, 56, 58, 60, 62, 67, 70, 72, 74, 77, 83
3	15, 16, 20, 25, 31, 54, 60, 64, 67, 72, 74, 77
4	5, 22, 44
5	2. 8, 19, 85
6	1, 4, 17, 28, 35, 50, 66, 79
7	1, 4, 17, 28, 50, 79
8	6, 10, 18
9	3
10	3, 7, 13, 24, 27, 30, 37, 55, 59, 68, 73, 75, 82
11	53, 57, 61, 65, 71
12	12, 38, 39, 40, 46, 48, 80, 87

APPENDIX C FIGURE #	WEAPONS REPLACEABLE ASSEMBLY (Cont)
13	11, 12
14	88, 89, 91
15	9, 21, 29, 36, 51, 63, 69, 84, 92, 94
16	14, 23, 26, 32, 34, 41, 42, 43, 45, 47, 49, 52, 76, 78, 81, 86, 90, 93, 95
17	33

The plots are in one of two forms, depending on the type of propulsion:

- PSD (g^2/Hz) versus frequency for the three jet aircraft
- acceleration (g) peak versus frequency for the turboprop aircraft

Each plot indicates the fuselage station (F.S.) measured in inches back from the nose of the aircraft, at which the measurement was taken. The axis of measurement for each data point is also identified. Ground take-off measurements are identified for those situations where they are larger than comparable flight measurements.

As indicated previously there was no new flight test data acquired for this study. The intent was to match existing vibration data at various aircraft equipment locations, nearest those WRA's selected. Since the aircraft examined in this study contains a large number of equipment because of their specific type mission, a major portion of these aircraft are compartmentized. Thus it was possible in some cases to utilize one measurement to describe the vibration environment of several WRA's. Therefore, there are situations herein, where several boxes are described by one figure, while is some cases there is one figure defining the vibration environment on only one WRA.

Some observation that were made after a review of this data are as follows.

For aircraft #1, the engine exhaust, inpinging on the fuselage, produced relatively severe vibration levels in the aft fuselage. Utilizing the 3.3 Hz. bandwidth filters, the data disclosed narrow band random peaks at structural modes (e.g., fuselage vertical bending and torsion) and

engine rotation speeds superimposed on a low level broad band random base.

Aircraft #2 which is powered by two turbojet engines located at the mid fuselage, has the equipment of interest for this study forward of the engines. The vibration environment is a low level broad band random spectrum with narrow band spikes associated with engine operational rotational speeds, structural modes and other major accessory drive systems.

The main power source for aircraft #3 is two turbofan engines with afterburner, mounted in the aft fuselage, with no fuselage exhaust impingement. The vibration environment was primarily low level broad band and narrow band peaks (associated with engine rotational speeds and structural modes) with the engine rotational vibration effects becoming less pronounced in the forward fuselage ahead of the engine. Higher levels were found nearer the engine area; however, equipment mounted between engines on the aircraft center line saw lower than expected levels due to structural attenuation and isolation.

The power sources for aircraft #4 are two wing mounted turboprop constant speed engines with four (4) bladed propeller. The environment was revealed to be sinusoidal with the highest vibration level at the propeller blade passage frequency of 73.0 Hertz (propeller blade passage frequency = NUMBER OF PROPELLER BLADES x ENGINE SHAFT FREQUENCY (Hz)), with lesser responses at various harmonics. The levels were recorded during both high speed cruise and ground engine run conditions.

Demonstration/Field Comparisons

Table 1 of Appendix C presents a comparison of field and laboratory vibration type and duration of exposure. The durations of exposure to permit a comparison of field and laboratory hours were derived as follows:

- Field Duration -- Field duration was defined as the number of hours a WRA would be exposed to vibration if it accrued the number of operating hours in the field equal to the demonstration test operating hours. As part of this study the ratio of operating time to flight time (O.T./F.T.) was determined for each WRA. Since the WRA is continuously exposed to vibration during flight, the duration was determined by dividing total demonstration test operating time by O.T./F.T.

- Laboratory Duration -- During the tests, vibration was applied for a fixed percentage of equipment operating time per test cycle. The total number of test operating hours was then multiplied by this percentage to establish the WRA vibration test hours.

In all cases, the duration of exposure to vibration was greater in the field than during the test (for equivalent operating hours) and in the majority of cases the differences in time was significant. Since the majority of the aircraft studied were gas turbine jet propelled, random vibration predominated as the mode of field excitation. All laboratory tests were performed under sine conditions.

3.4.3 Moisture

Moisture as defined for this study was a somewhat all inclusive term encompassing humidity, precipitation, condensation, salt fog, icing, etc. Although all forms of moisture are present and may effect equipment performance, it is the absolute humidity (mass of water vapor per unit volume of space) manifested as condensation and precipitation which most seriously affect electronic equipment. For example, for icing to occur requires a particular set of circumstances:

- Soak of aircraft at low temperature (due to high altitude flight) and then a descent at some optimum rate to drop surface temperature below dewpoint, or

- Flight of aircraft through a supercooled cloud which upsets the equilibrium of the unstable air mass causing freezing of water droplets.

Even if icing did occur, it is limited to external aircraft surfaces and does not directly affect internal avionic equipment. More serious forms of moisture exposure may occur in three basic areas:

- Water vapor migration and penetration due to vapor pressure differences between the compartment ambient and box interior.

- Prolonged exposure to high ambient humidity (on the ground) manifested as free moisture:

  - Carrier operation which would include a certain salt content and result from spray.

  - Night temperature dropping below the dewpoint causing precipitation.

- Freezing of entrapped moisture at a certain altitude.

- Condensation, again due to altitude (hence temperature) variations, within forced air cooled boxes.

In forced air cooled boxes any free moisture which may be present would probably tend to degrade electronic equipment performance. However, if the equipment were cold plate cooled, air would not impinge on the components and moisture would actually provide additional cooling represented by the latent heat of the water.

Many laboratory demonstration requirements dictate that extreme precautions be taken so that humidity effects do not adversely affect equipment performance.* It is estimated that the relative humidity maintained during a typical test is less than 10%. While it is difficult to quantify moisture due to field operation, certain facts are known. The world average is 75% and the aircraft in this study are exposed to at least this condition for a good percentage of their life. In addition, carrier operations add considerable free moisture including a certain salt content.

3.4.4 Altitude

All selected study equipments are exposed to some low pressure condition during flight. Cockpit equipment, however, is maintained at ambient pressures up to a given altitude (generally 8,000 feet) and then exposed to a controlled pressure (higher than ambient) up to the operational ceiling of the aircraft.

The effects of altitude are manifested in two ways:

- Steady state exposure

- Cycling effects due to aircraft altitude variations

Grumman experience indicates that the steady state condition generally causes problems due to a disruptive electric discharge (arc-over), causing serious damage to electronic components. The cycling conditions result in failures of gasket sealed enclosures permitting loss of gas or fluid and entry of moisture. Thus the major effect of pressure variation on avionics

*Stated in reference 4 and 5 and implicit in the temperature profiles of reference 3.

equipment is structural rather than thermal.

Analysis and flight test data on avionics equipment indicate that while decreased pressure results in some loss of cooling capability, the effect is minimal. While it is apparent that reduced pressure has little effect on the cooling of a forced cooled box, the situation of an ambient cooled unit must be investigated in more detail. The general contention is that the temperature difference between a component within the box and the ambient is not significantly affected by reducing pressure. The heat balance on an ambient cooled box can be stated as follows:

$$Q_D = Q_C + Q_R$$

where

Q_D = power dissipation of box

Q_C = total convective cooling from the box case

Q_R = total radiant cooling from the box case

Since the primary mode of heat transfer from the component to the case is through conduction, it is obvious that if the case temperature changes a given amount, the component will change an equal amount since conduction through the cards in the box is independent of pressure. Thus it remains to be shown that the temperature difference between the case and ambient is independent of pressure. Analyzing the heat balance equation further, it can be noted that

$$Q_C = hA\,(T_{case} - T_{Ambient})$$

where

h = heat transfer coefficient (a function of ambient pressure)

A = box convective area

and

$$Q_R = \tau\,\epsilon\,(T_{case}{}^4 - T_{Ambient}{}^4)$$

where

τ = radiation constant

ϵ = case emissivity

Generally, the heat transfer coefficient is a function of pressure to a power of approximately $\frac{1}{2}$. Radiation cooling, however, is independent of pressure. It can be observed that despite the fact that a reduction in

pressure causes a change in temperature difference between case and ambient, the radiant cooling will be increased as a function of the fourth power of the case temperature. Thus the resulting change in temperature difference between case and ambient is small.

In addition, the ambient temperature decreases with altitude. This causes the term "$T_{case} - T_{ambient}$" in the above equations to increase thus permitting more heat to be transferred by radiation and convection. This also compensates for the pressure effect.

As an example, consider a typical ambient cooled box with control panel. The box is approximately 6" x 8.4" x 2.6" and the attached control panel is 9" x 4.1". The unit dissipates a total of 25 watts. Thus the dissipation per unit surface area is approximately 19 W/ft.^{2}, which is typical of the boxes considered in this study. At sea level the temperature difference between the case and ambient is 36°F. The difference is 41°F at 30,000 feet, 44°F at 50,000 feet, and 47°F at 70,000 feet.

Figure 14 is a plot of a typical component temperature profile during flight. It also shows that an increase in altitude does not result in a significant increase in the temperature difference between the component temperature and the compartment. Additionally, for fan cooled avionics (self contained blower), current design practices employ high-slip motors which result in a relatively constant mass flow rate of ambient air over internal components. The constant mass flow rate effectively negates the loss of cooling capability due to decreased pressure. Table B-3 presents the field levels and rates of change for each equipment including the differences between field and laboratory. Since all laboratory testing was performed under sea level conditions (14.7 PSI), field conditions were lower in each case.

3.4.5 Input Voltage

All electric power, produced by on-board generating systems and supplied to airborne equipment at the equipment terminals, is controlled in accordance with the requirements of MIL-STD-704A (ref. 9). Discussions with engineering and flight test personnel, and actual measurements made of input voltage confirm the fact that those values are within the limits

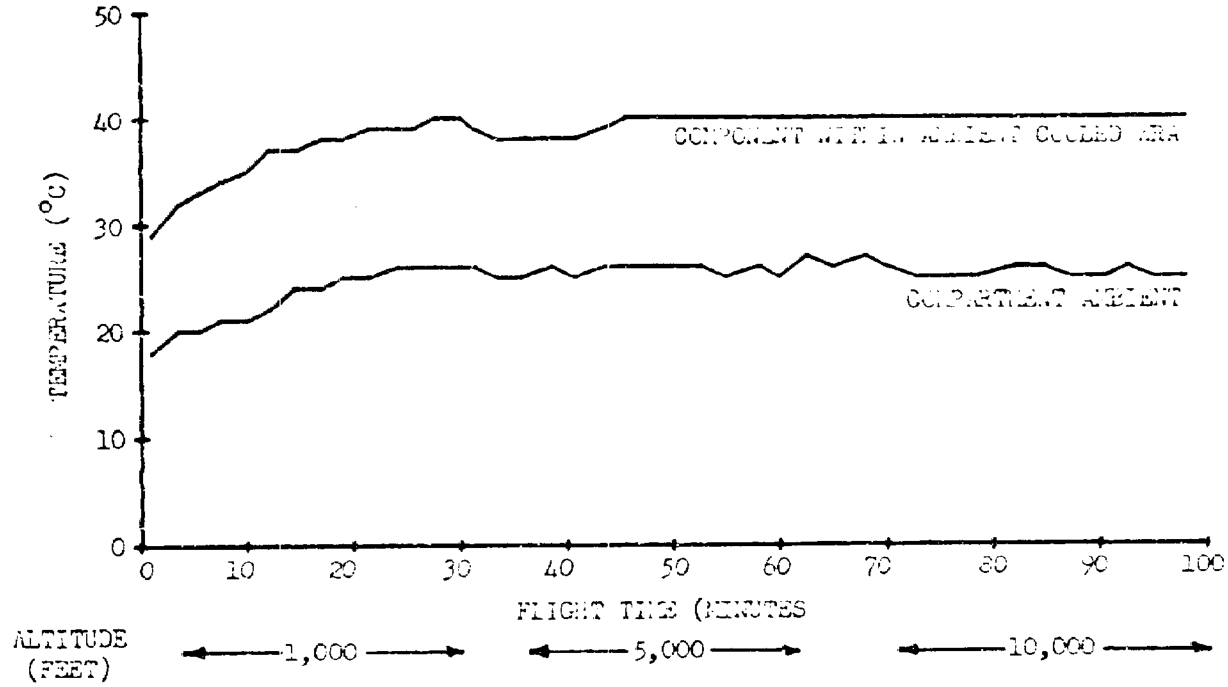


FIGURE 14 EFFECT OF ALTITUDE ON COMPONENT TEMPERATURE

prescribed by MIL-STD-704A. A review of the available field data indicates that there were no identifiable problems directly attributable to transient or steady state voltage levels or frequency variation.

Normally, voltage regulation for modern aircraft power systems (A.C. generating systems) is typically within 3% of nominal, 99% of the time. Accordingly, DC voltages derived from these sources would have a similar variation. Such variations should have a negligible effect on equipment reliability. Consequently, the ± 10% variation in supply voltage specified in Para. 5.2.4 of MIL-STD-781B is not realistic. Since the intent is to demonstrate the reliability of typical equipment under typical service conditions, it is recommended that the reliability demonstration test be performed under nominal voltage conditions.

Under special situations, e.g., new equipment on old vintage A/C (D.C. generating systems), it is possible that due to generator capacity/ cable sizing the equipment input voltage could be at the low end of the tolerances specified in MIL-STD-704, Tables I and II most of the time, instead of nominal. However, such a situation should be considered a special case and should be considered in the equipment specification.

Section IV

MTBF Analysis

4.1 Approach

This section describes the results of the analyses performed to determine the demonstrated and field MTBF values for each selected WRA. The major thrust of this effort was to define and then determine a consistent set of values that could be legitimately compared and subsequently used for further analyses. Assuring consistency between laboratory and field values was vital since a fundamental study objective was to determine if environmental differences between the laboratory and field contributed to apparent reliability differences. Thus, care must be taken to assure that any reliability difference noted was "real" and not just the result of differences in definition, groundrule, measurement technique, reporting system, etc. Laboratory test reports were the source of data for determining demonstrated MTBF. The primary source of data for field MTBF determination was the Navy's 3M system supplemented by field service reports, project reliability reports, or development flight test failure reports.

4.2 Demonstration

4.2.1 WRA θ Derivation

Realization that WRA's of the same subsystem (equipment) are not necessarily co-located within the aircraft and may therefore see different environmental exposure in the field, plus the desire to increase the data base to provide greater statistical validity, were the motivating factors for performing the study at the WRA level rather than at the equipment level. This generally proved to be no problem since most of the data required for the study, both in terms of environments or field failure experience, was available at the WRA level. The one area that created some difficulty was the determination of a demonstrated reliability value for each WRA. Inasmuch as the reliability demonstration tests were run at the equipment level, the test parameters (i.e., test time, allowable number of failures) were determined by the equipment's reliability requirements. Thus each constituent WRA of an equipment accrued sufficient operating time and failures to permit demonstra-

tion and estimation of the equipment MTBF. However, estimation of a WRA MTBF from this same test experience by conventional statistical methods often yielded unreasonable results.

To illustrate this point, consider the following hypothetical example. Assume an equipment is composed of three WRA's: A, B, and C. The reliability requirement of the equipment is an MTBF of 100 hours and the predicted MTBF's of the WRA's are 110, 10,000 and 10,000 hours, respectively. Assume that the equipment had accrued 289 test hours with one failure charged to WRA A, thus successfully passing the requirements of MIL-STD-781 Test Plan III. Thus the available test information for MTBF estimation is as follows:

WRA A	WRA B	WRA C
289 test hours	289 test hours	289 test hours
1 failure	0 failure	0 failure

MTBF point estimates are conventionally determined from:

MTBF = Test Time/Number of Failures

For the cases where zero failures were charged, the convention adopted was to use the 50% confidence estimate. This is determined for

$$\text{MTBF} = \frac{2 \cdot T}{x^2_{(2,.5)}}$$

where T = test time

$x^2_{(2,.5)}$ = is the tabulated 50th percentile of the chi square distribution for 2 degrees of freedom

Thus, for the above example, the MTBF estimates for each WRA are:

A = 289 hours

B = 417 hours

C = 417 hours

The unreasonableness and lack of appeal of this method is evident when one compares these estimates with the MTBF predictions. Whereas the relative difference between WRA B and WRA A predictions are approximately 100:1, the estimates based on test results are less than 2:1. This dramatic change in relative value cannot be explained by any unusual test results since WRA B did not experience any test failures. This difference is attributed to the

fact that WRA B was not tested for a sufficient length of time to obtain a valid estimate of its MTBF. This method becomes even more unreasonable if one assumes that there were no failures during the demonstration test. In this case, each WRA, regardless of the obvious differences in complexity, failure potential, etc., would exhibit the same MTBF test estimate.

Two approaches were considered as possible methods for circumventing this problem as follows:

Bayesian Updating -- This method, based on an application of Bayes' Theorem, has been popular in recent years. It permits the analyst to "mix" test results with prior information/history to arrive at a current estimate. The underlying theory and derivations are well documented in the literature (ref. 10). The major applicable points are identified below:

1. Assume that the MTBF (Θ) is a random variable with probability density function $f(\Theta)$

2. Assume $f(\Theta)$ to be an inverted gamma distribution

3. Assume a prior distribution of Θ exists with parameters T and r (where T, accumulated test time, and r, the number of failures, are generally based on previous testing). The mean of this distribution is $T/(r-1)$.

4. Given additional testing of t hours and k failures, the posterior distribution of Θ is again an inverted gamma with parameters $T + t$, $r + k$ and the mean of this distribution is $(T + t)/(r + k - 1)$.

Since, in this study application, no previous reliability testing was performed prior to the demonstration test, several assumptions concerning the parameters of the prior distribution of Θ would have to be made.

It was assumed that the predicted value of Θ would serve as the mean of the distribution. The rationale for selecting the predicted value was:

a) At the time of the demonstration test this value represented the "state of knowledge" of the WRA's MTBF.

b) One can reasonably assume that the prediction of the reliability of the test article satisfied the reliability requirement imposed on the WRA, in that a satisfactory prediction is generally a precursor to the demonstration

test and that "reliability fixes" indicated by the prediction analysis as necessary to achieve the requirement had been incorporated in the test article.

Once a value for the mean is assumed and since the mean = $T/(r - 1)$, knowledge of either T or r (neither of which exist in fact since no prior reliability testing was performed) would permit determination of the other. It was more appealing logically to make assumptions concerning T (analog of test time) rather than $(r - 1)$ (analog of number of prior failures). A desirable characteristic of the parameter used is, that it be a measure of the degree of belief one has, at the time of demonstration testing, that the prediction is the true MTBF of the WRA. Thus, it is somehow related to experience. The parameter T (test time analog) more closely satisfies this requirement since generally, the more testing done, the more one knows about the design.

The final assumption necessary for use of this approach is the selection of a value of T. Desirable attributes of the selected value include:

a) It should be small enough in magnitude to give reasonable weight to the actual demonstration test results.

b) It should be large enough in magnitude that some degree of stability in the estimation process has been achieved (i.e., small changes in T do not produce large changes in the posterior value of the mean).

c) It should be less than, yet relatable to all the development testing that preceded the reliability test. Though, in a strict sense, these tests are not reliability oriented, they each in turn (assuming that knowledge gained is incorporated in the design), provide additional assurance that the unit will function properly in its intended use enviroment.

This approach was abandoned because the predictions available for use were either those originally performed during the degree phase or those performed during this study using the coordination copy of MIL-HDBK-217B as the common source. Neither source of reliability prediction data was considered acceptable for use in this application. For the former, since the prediction for each WRA was performed at a different time, there was no assurance that any consistency in failure rate source, groundrule, etc., was present from WRA to WRA. For the latter,

although consistency could be guaranteed, since they were all performed under the aegis of this study, significant differences in failure rates between the advance version of MIL-HDBK-217B used and that ultimately released precluded use of these values in an absolute sense. Insufficient time remained from the time of release of MIL-HDBK-217B to the scheduled completion of the study to repredict all the WRA MTBF's using the final issue as the source document. Furthermore, the potential weakness in using any prediction technique as an absolute value suggested that a relative allocation would be less sensitive to variations of the data base.

<u>Proportional Allocation</u> -- This approach is the one that was ultimately adopted for use in the study. It assigns as a WRA's demonstrated failure rate that same proportion of the equipment's failure rate as the WRA prediction is to the equipment prediction. Thus, for example, if an equipment is composed of three WRA's with predicted values:

WRA A	WRA B	WRA C
Predicted MTBF = 200	Predicted MTBF = 1000	Predicted MTBF = 10,000
Predicted F.R. = .005	Predicted F.R. = .001	Predicted F.R. = .0001

The equipment failure rate (F.R.) is then approximated as .005 + .001 + .0001 = .0061 and each WRA's proportional share is

WRA A = .005/.0061 = 81.97%
WRA B = .001/.0061 = 16.39%
WRA C = .0001/.0061 = 1.64%

Thus, if the equipment accrued 540 hours with two failures during demonstration testing, the estimated demonstrated equipment failure rate is 2/540 = .0037037. This procedure would then assign to each WRA as a demonstrated failure rate value the following:

WRA A -- .8197 x .0037037 = .003036
WRA B -- .1639 x .0037037 = .000607
WRA C -- .0164 x .0037037 = .000061

with corresponding demonstrated MTBF values

WRA A -- 329

WRA B -- 1647

WRA C -- 16463

This method eliminated some of the objections with the previously described approaches in that it yielded values which have some relationship to known differences in complexity among the WRA's while circumventing the necessity for directly using predictions that are questionable.

Implicit in the use of this approach is the assumption that relative differences among the WRA's that constitute an equipment are generally preserved when going from a prediction to a demonstration. Though this assertion cannot be proved directly, the following analysis is offered to justify this approach. Inasmuch as field MTBF's for each WRA were determined for this study, the relative contribution of each WRA to its equipment field failure rate could readily be obtained and compared to its corresponding relative contribution to the equipment prediction. The rationale for this is, if the relative contribution of a WRA failure rate to its equipment failure rate remains essentially the same from prediction to the field (considering all the factors influencing the field value), then it is reasonable to assume that the relative contribution comparing prediction to demonstration (where all these factors are not present) is also the same. The table below shows the distribution of the difference between field and prediction contribution.

DISTRIBUTION OF DIFFERENCES IN WRA RELATIVE CONTRIBUTION

Difference*	Frequency	% of Observation
Less than .10	67	75
.10 - .19	11	12
.20 - .29	5	6
.30 - .39	4	4
.40 and greater	3	3
	90	100

$$^{*}\text{Difference} = \left| \frac{\text{WRA } \lambda_{\text{FIELD}}}{\text{EQUIPMENT } \lambda_{\text{FIELD}}} - \frac{\text{WRA } \lambda_{\text{PREDICTED}}}{\text{EQUIPMENT } \lambda_{\text{PREDICTED}}} \right|$$

Note that there are only 90 observations because there were five single WRA

equipments in the study. As the table indicates, there was less than a .10 difference in relative contribution in 75% of the WRA's included. This large number of similar values tends to support the above assertion, i.e., the ratio, WRA λ (pred)/Equip λ (pred), approximates WRA λ (Field)/Equip λ (Field).

As a further precaution, an analysis was performed to determine if the differences between field and prediction contribution were related to "part mix." Two measures related to part population characteristics were analyzed: % microcircuits and % TX or better. These two measures were selected because they presumably represented the areas of greatest change between the coordination copy and the released version of MIL-HDBK-217B. The 90 WRA's were partitioned into two subpopulations -- those with differences less than 0.10 and those with differences of 0.10 and greater. The distribution of each subpopulation against each part population measure was determined and summarized in Tables 8 and 9 below. Examination of Table 8 indicates that there is no apparent difference in microcircuit complement distribution between the two subpopulations. Approximately three quarters of the WRA's in each group have less then 20% microcircuits. The remaining quarter is approximately evenly divided between the 20 - 50% and the over 50% group.

Inspection of the data presented in Table 9 appears to indicate that the two groups have somewhat different distributions and that the higher reliability parts may be affected differently in the field than the prediction method assumes. Specifically, the data appears to indicate that there is a tendency for WRA's with a large percentage of TX parts to have greater differences (39% for greater than 0.10 versus 15% for less than 0.10). A more detailed analysis of the data was performed to determine if this, in fact, was the case. The group where the differences were greater than 0.10 was further partitioned into:

- those where the field exceeded the prediction by 0.10
- those where the prediction exceeded the field by 0.10

and the distribution of each against the % TX or better was determined. If the Hi-Rel parts were affected in the field differently than parts of lower quality, one would expect the two resulting distributions to be completely dissimilar. The results could be observed in one distribution being heavily

TABLE 8 DISTRIBUTION OF DIFFERENCES* IN WRA RELATIVE CONTRIBUTION VS. % MICROCIRCUITS

% MICROCIRCUITS	DIFFERENCES* GREATER THAN 0.10		DIFFERENCES* LESS THAN 0.10		TOTAL POPULATION	
	FREQUENCY	%	FREQUENCY	%	FREQUENCY	%
Under 20%	17	74	48	72	65	72
20 to 50%	3	13	11	16	14	16
Greater than 50%	3	13	8	12	11	12
TOTAL	23	100	67	100	90	100

TABLE 9 DISTRIBUTION OF DIFFERENCES* IN WRA RELATIVE CONTRIBUTION VS. % TX OR BETTER

% TX OR BETTER	DIFFERENCES* GREATER THAN 0.10		DIFFERENCES* LESS THAN 0.10		TOTAL POPULATION	
	FREQUENCY	%	FREQUENCY	%	FREQUENCY	%
Under 20%	10	44	45	67	55	61
20 to 50%	4	17	12	18	16	18
Greater than 50%	9	39	10	15	19	21
TOTAL	23	100	67	100	90	100

$$* \text{Difference} = \left| \frac{\text{WRA } \lambda_{\text{Predicted}}}{\text{Equipment } \lambda_{\text{Predicted}}} - \frac{\text{WRA } \lambda_{\text{Field}}}{\text{Equipment } \lambda_{\text{Field}}} \right|$$

weighted toward the under 20% interval while the other distribution would be heavily weighted towards the greater than 50% interval. The two distributions are presented in Table 10 and show a great similarity between the two, thus negating any causative relationship.

The above analyses, taken together, suggest that:

- the differences between field relative contribution and predicted relative contribution are small, and
- those differences that are large cannot be associated with diffrences in part type mix.

The conclusion drawn from the above is that the ratio of a WRA failure rate to its corresponding equipment failure rate is essentially preserved from prediction to the field.

4.2.2 Test Ground Rules

Although the major study emphasis is directed toward environmental differences, it was also recognized that other factors, including test ground rules, might also contribute significantly to differences in MTBF. During the reliability demonstration tests performed on the selected equipment, definitions per MIL-STD-721 (ref. 11) and ground rules specified in MIL-STD-781 and AR-34 (ref. 12) were generally employed. These were modified and/or supplemented in varying degree for each of the equipment tests performed. Failure scoring criteria for field and laboratory must therefore be compatible and a realistic set of ground rules must be defined. During this study, all ground rules and definitions were reviewed, including those appearing in AR-34, MIL-STD-781, MIL-STD-721, and those peculiar to each of the study equipment tests. The rules obtained from these sources were then screened and either rejected, used as-is, modified, or supplemented. The set of "standard" rules established was then applied to each of the demonstration tests resulting in reclassified failures and revised θ's. This review was based on the premise that field and laboratory θ's must be derived in an identical manner using the same scoring criteria for failure classification. The following rules were established and used to classify failures:

FAILURE

A failure is the cessation of equipment operation or an out-of-

TABLE 10 DISTRIBUTION OF DIFFERENCES IN CONTRIBUTION GREATER THAN 0.10 VS. % TX OR BETTER

% TX OR BETTER	FIELD - PREDICTION GREATER THAN 0.10		PREDICTION - FIELD GREATER THAN 0.10	
	FREQUENCY	%	FREQUENCY	%
Under 20%	6	46	4	40
20 to 50%	2	15	2	20
Greater than 50%	5	39	4	40
TOTAL	13	100	10	100

specification condition of a performance characteristic at any environmental condition within the specified limits.

RELEVANT FAILURES

All failures are relevant unless determined by the procuring activity (or an authorized representative thereof) to be caused by a condition external to the equipment under test which is not a test requirement. Relevant failures include:

- Design/Workmanship Failures: Failures due to design deficiencies or poor workmanship of either the equipment or component parts.

- Component Part Failures: Failures due to defective component parts shall be classified as relevant failures. In the event that several component parts of the same type fail during the test, each one shall be considered a separate relevant failure, unless it can be shown that one failure caused one or more of the others.

- Wearout Parts: Certain parts of known limited life, such as batteries, may have a life stipulated prior to the initiation of testing as approved by the procuring activity. Failures of these parts occurring prior to the end of the stipulated period are relevant. Failures of these parts occurring after the stipulated period are nonrelevant, but any dependent failures caused thereby are relevant.

- Multiple Failures: In the event simultaneous part failures occur, the entire incident shall be counted as one relevant failure. (Since multiple failures cannot be distinguished from dependent failures in the field data, this ground rule was included to assure consistency with the field analysis.)

- Intermittent Failures: The first occurrence of an intermittent failure on any one equipment shall be counted as a relevant failure. Subsequent occurrences of the same intermittency on that same unit will be considered nonrelevant provided that reasonable effort was made to assure that a failure condition no longer existed.

- Adjustments:

  - Anticipation of failure shall not be justification for preventive maintenance, i.e., if an output is observed to be degrading but is still within specification limits. No replacement is permitted and any adjustment of a control is a relevant failure unless both the control and the indicator

signaling impending failure are an integral part of the equipment under test and are available and accessible to the aircrew during normal flight operation.

- Inaccessible Controls: Each adjustment of a control which is inaccessible to the operator during normal use is a relevant failure.

- Failures of Built-in-Test: Any malfunction (including a false alarm) of the Built-in-Test features of the equipment shall be classified as a relavant failure.

- When a failure occurs and a change is incorporated (design, part or process) which will correct the problem, the first occurrence shall be scored relevant and all subsequent failures of the same type occurring during the test but before corrective action has been incorporated, shall be non-relevant.

- Failures detected during the final functional test following the successful completion of the test shall be scored relevant, if the equipment used to monitor the performance characteristics during the demonstration was not capable of detecting that failure.

NON-RELEVANT FAILURES

Only those failures listed below may be counted as non-relevant.

- Failures directly attributable to improper installation in the test chamber.

- Failures of test instrumentation or monitoring equipment (other than the Built-in-Test function).

- Failures resulting from test operator error in setting up, or in testing the equipment.

- Dependent failures, unless caused by degradation of items of known limited life. (At least one relevant failure shall be counted when a dependent failure is claimed.)

- Failures attributable to an error in the test procedures.

- The second (and any subsequent) occurrence of the same intermittent failure on the same unit.

- Failures occurring during burn-in.

- Failures occurring during test "down time."

- Malfunctions of the Time Totalizing Meters.

- Failures clearly attributable to an overstress condition in excess of the design requirements.

EXCLUSIONS

A failure, classified as relevant, may be considered to be non-relevant for this study provided that all of the following conditions are met:

- Corrective action (an equipment design, part, or production process change) has been made in accordance with the applicable reliability test specification or standard on all equipment of the lot from which the reliability test sample was drawn, and;

- Sufficient test data has been accumulated to indicate the corrective action is effective in eliminating the failure mode, and;

- Approval of the procuring activity (or authorized representative) is obtained for exclusion of the failure.

NOTE: The first occurrence of such failure shall be scored relevant.

Table 1 of Appendix D shows the WRA MTBF's demonstrated under the ground rules and scoring criteria in effect at the time of testing. Revised values after reclassifying failures with the above ground rules are also presented.

The frequency distribution of MTBF values for each situation was determined and some summary statistics evaluated to gain some gross insight into the impact of this reclassification of the original source data. Figure 15 presents the frequency polygon for both before and after reclassification. Some pertinent statistics for each polygon are shown below.

SUMMARY OF DEMONSTRATION TEST DATA BEFORE AND AFTER RECLASSIFICATION

	MTBF (Hours)		
Statistic	Before	After	% Change
Mean	16377	10213	38
Standard Deviation	32438	26073	20
Median	2150	1550	28
25th Percentile	940	560	40
75th Percentile	13000	7900	39

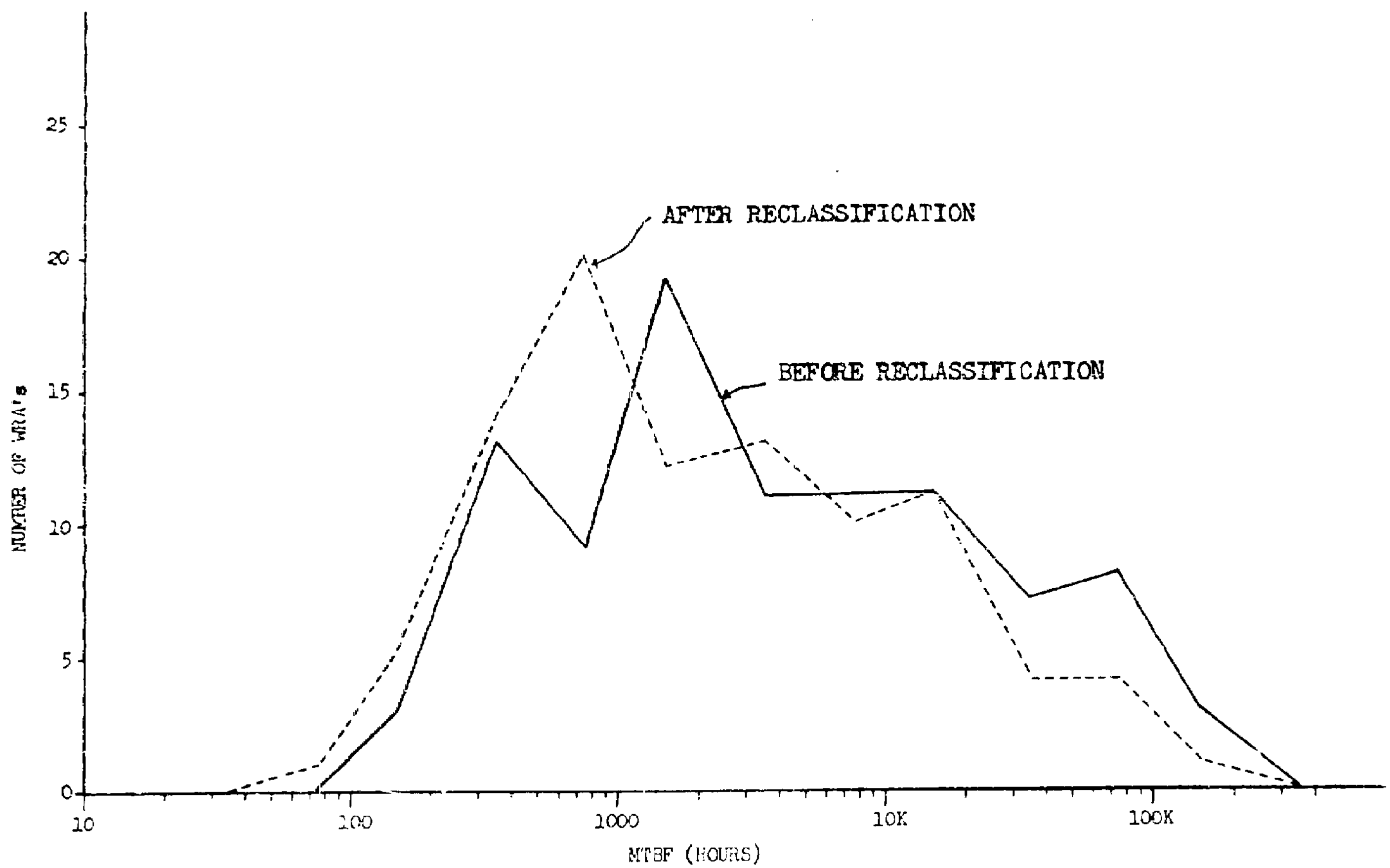


FIGURE 15 IMPACT OF RECLASSIFICATION ON THE DISTRIBUTION OF DEMONSTRATED MTBF'S

A review of the Figure and Table indicates a general similarity in shape of the two distributions with the reclassification producing a 30-40% downward shift in each of the location statistics.

4.3 Field

Consistency in failure definition and time measurement is necessary, to assure that field-laboratory reliability comparisons are valid. Estimates of MTBF in the field should be determined from equipment operating time and the number of equipment failures experienced during that operating time. All too often, because of inability to discriminate in data, field MTBF's are reported as the resulting quotient after dividing aircraft flight time by the total number of maintenance actions reported. This measure, referred to as Mean Flight Time Between Maintenance Action (MFTBMA), though perhaps having value for Operations personnel for planning and resource control, is not comparable with either predicted or demonstrated MTBF's. Whereas, the MFTBMA counts all incidents (i.e., failures, false alarms, preventive maintenance, cannibalization, induced failures, etc.), individual program demonstration test ground rules exclude every type of incident except certain failures. It has not been uncommon for test ground rules to exclude some legitimate failures as well. In addition to failure count definition not being compatible, the substitution of aircraft flight time for equipment operating time creates artificial differences in field-demonstration reliability comparisons. Flight hours may differ from equipment operating hours by such factors as duty cycle, ground operating time while installed in an aircraft, and bench time.

Thus, both numerator and denominator of an MFTBMA does not relate to the usual definition of MTBF. Except for those items that have a very short duty cycle, equipment operating time is longer than flight time. Usually the number of equipment failures is less than the number of maintenance actions. Therefore, MFTBMA is less, often much less, than MTBF. And if, as indicated previously, test ground rules have eliminated some relevant failures, as derived for this study, from the count, then the disparity between laboratory demonstrated and field measured reliability is even greater.

The analyses that were performed on the field data to bring it to a common basis with the laboratory included evaluating equipment operating time and determining the number of equipment failures in accordance with ground rules

established to be comparable to those established for the laboratory MTBF analysis.

4.3.1 Equipment Operating Time

As indicated, equipment operating hours is the proper measure of time for determining an MTBF. Aircraft flight hours, however, are frequently used as the time base for MTBF determination. The use of this parameter has become so widespread that it has essentially become the conventional one to use in reporting field reliabilities from existing military data systems. Availability of data is the predominant reason for using this parameter. As a matter of course, the military data systems currently in use collect, summarize, and report aircraft flight hours. This information is thus readily available to all users, both military and contractor. Collection of actual equipment operating times, however, is much more difficult and so much less regular. Elapsed Time Indicators (ETI's), wherever installed, would have to be read periodically or the turn-on/turn-off times for each equipment without an ETI would have to be determined for each WRA installed in every aircraft in the inventory. Because of the expensive nature of such an undertaking, this effort is not generally done.

The net effect of using flight times instead of operating hours is to, everything else being equal, understate the MTBF value for those equipments that accrue relatively large amounts of ground time while installed in the aircraft and to overstate the MTBF value for those equipments that have a relatively short duty cycle (discounting that there is no present, practical method of assessing the impact of the environments on non-operating equipments).

Since equipment operating time was the proper parameter to use in this study, an analysis was performed to determine operating time from the information readily available in the data system, viz., aircraft flight times.

Maintenance personnel are required to note the serial number and ETI reading on every item removed from or installed in an aircraft. Although this requirement is not rigidly enforced, it is adhered to sufficiently to provide useful data for analysis. The analysis goal was to develop a factor which when multiplied by aircraft flight hours would yield WRA operating hours. A sample of data points was assembled for each WRA where each data point represented the difference in ETI readings between installation and removal of a given serial

number from a particular aircraft. Since the dates of installation and removal were given, the corresponding aircraft flight time between installation and removal dates was noted for each ETI difference. The ratio of total ETI differences to the total corresponding flight time was determined for most WRA's in this manner. For any item where insufficient data existed to develop its own ratio, the ratio of one whose operating profile most closely resembled it was used. Each such ratio multiplied by total aircraft flight time became the estimate of total WRA operating time.

4.3.2 Failure Assessment and Ground Rules

Field failure criteria were developed in a manner similar to and compatible with those developed for the reliability demonstration tests. All reported incidents were considered relevant and counted unless the equipment was:

- bad from supply
- removed for preventive maintenance
- removed for the convenience of the maintenance crew to gain access to another equipment
- removed from an aircraft and not verified bad in the shop
- removed for modification
- damaged as a result of abuse, combat, mishandling, etc.
- part of a cannibalization action

Flight discovered anomalies that could not be verified by the ground crew while the WRA was still installed in the aircraft were at first considered to be non-relevant. After a preliminary review of the data and due consideration of the operational environment, it was decided to include these incidents in the failure count. The rationale for this decision was based on the realization that the flight environment and ground environment are different. Thus it is possible, and previous experience corroborates this as happening, for equipment malfunctions to be observed in flight and then "disappear" on the ground. The normal procedure in the field is to remove a WRA from the aircraft only if the reported malfunction is verified by the ground crew. However, the ability to reproduce the flight environment on the ground is not available, specialized diagnostic equipment is generally not available at the aircraft level, and the level of diagnostic skills among maintenance personnel is lower

in the field than in the laboratory. Thus it is more likely in the field that report "false alarms" were, in fact, failures, and, therefore this type of incident was included in the failure count.

Only complaints recorded against a particular WRA were counted. The system level complaints were excluded from the failure count because experience indicated that these were generally more associated with integration and system/aircraft interface problems. It was also felt that any system level false alarm that was caused by faulty WRA performance during flight would persist from flight to flight and would ultimately be isolated to the appropriate WRA.

Two estimates of WRA field MTBF are presented in Table 2 of Appendix D. Each represents the results of a review of the equipment failure history for all of 1973 and, in some cases, the first quarter of 1974. The first estimate is the equipment MFTBMA and results from dividing aircraft flight time by total maintenance actions. The second measure is the resultant value after dividing estimated equipment operating time by the number of reclassified failures. As indicated, it is this second measure that is comparable to demonstration results. As with the case of the demonstration data, the distribution of the field MTBF for before and after reclassification was determined. The resulting frequency polygons are presented in Figure 16 and summary statistics are presented below.

SUMMARY OF FIELD DATA BEFORE AND AFTER RECLASSIFICATION

	MTBF (Hours)		
Statistic	Before	After	% Change
Mean	1072	1861	74
Standard Deviation	2214	3446	56
Median	255	510	100
25th Percentile	100	205	105
75th Percentile	700	1500	114

A review of this data indicates that reclassification resulted in a smoother distribution with the location statistics shifted upward by approximately 100%.

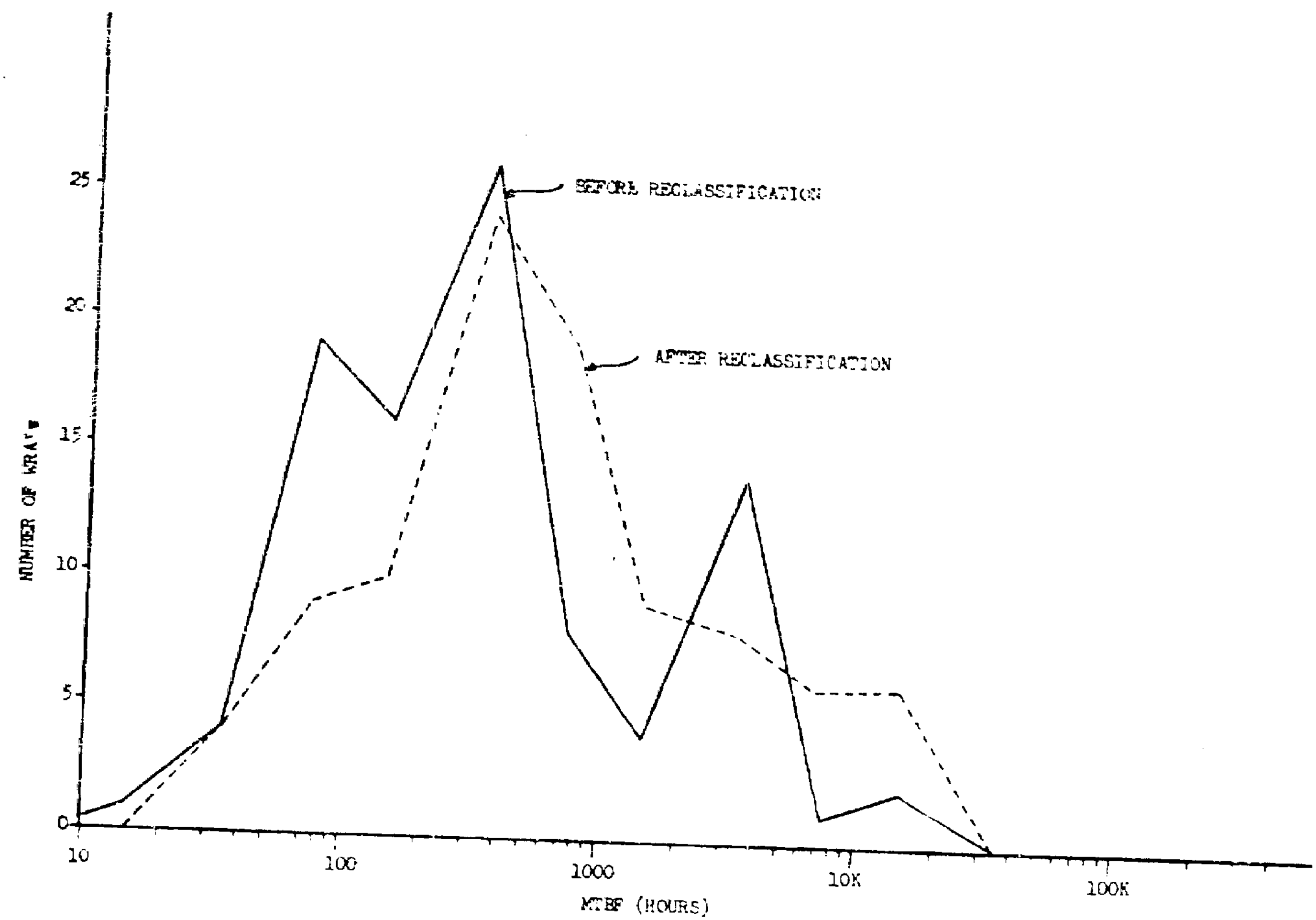


FIGURE 16 IMPACT OF RECLASSIFICATION ON THE DISTRIBUTION OF FIELD MTBF'S

4.4 Impact of Ground Rules

To gain some appreciation for the impact of reclassification on demonstration/field comparisons, the information presented in Figures 15 and 16 was replotted to readily show the differences in demonstration/field comparison when using "before" data versus using the "after" data. The results of this are shown in Figures 17 and 18. The corresponding comparison in summary statistics are shown below.

Examination of this information reveals the great dissimilarity in appearance and location statistics between the two "before reclassifying" distributions. The field peaks at a low MTBF and decreases rapidly in frequency whereas the distribution of demonstrated values appears flatter over a longer interval.

A comparison of the two "after" distributions indicates that the two are closer to each other than in the "before" case and have approximately the same shape. Comparison of location statistics reveals that in the "before" case the demonstration values are approximately an order of magnitude larger than the field. For the "after reclassification" situation, the demonstration values are 3-5 times greater than its corresponding field value.

SUMMARY COMPARISON OF DEMONSTRATION AND FIELD DATA BEFORE AND AFTER RECLASSIFICATION

STATISTIC	MTBF BEFORE RECLASSIFICATION		MTBF AFTER RECLASSIFICATION	
	Demo	Field	Demo	Field
Mean	16377	1072	10213	1861
Std. Deviation	32438	2214	26073	3446
Median	2150	255	1550	510
25th Percentile	940	100	560	205
75th Percentile	13000	700	7900	1500

This analysis indicates that though field and demonstration MTBF's are very different, the difference though large, is not as great as the "raw" data would indicate. These comparisons, however, were performed on resulting summary statistics from the two distributions. They therefore do not reflect the extent of the MTBF difference on individual WRA's, nor

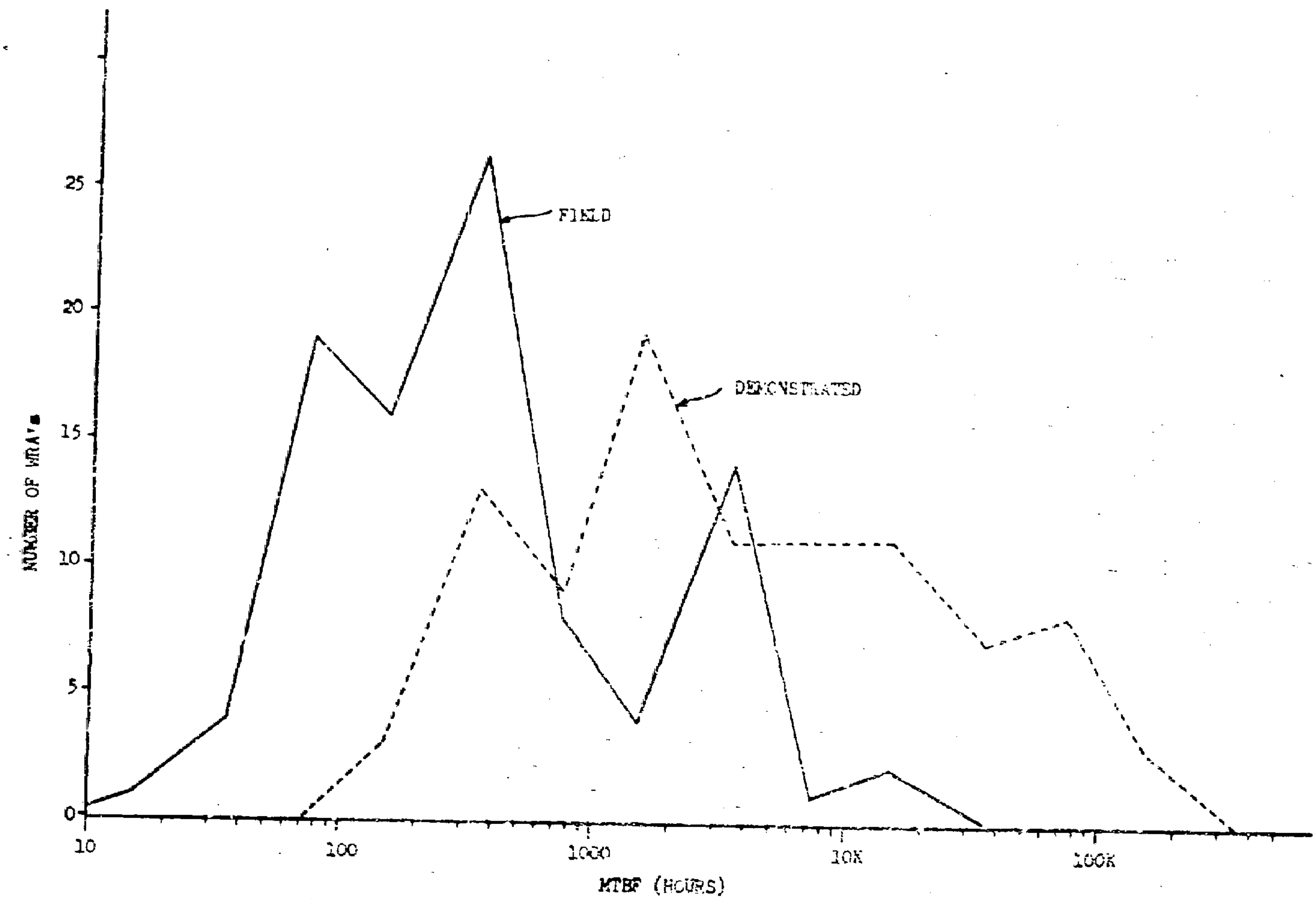


FIGURE 17 DISTRIBUTION OF DEMONSTRATED AND FIELD MTBF'S BEFORE RECLASSIFICATION

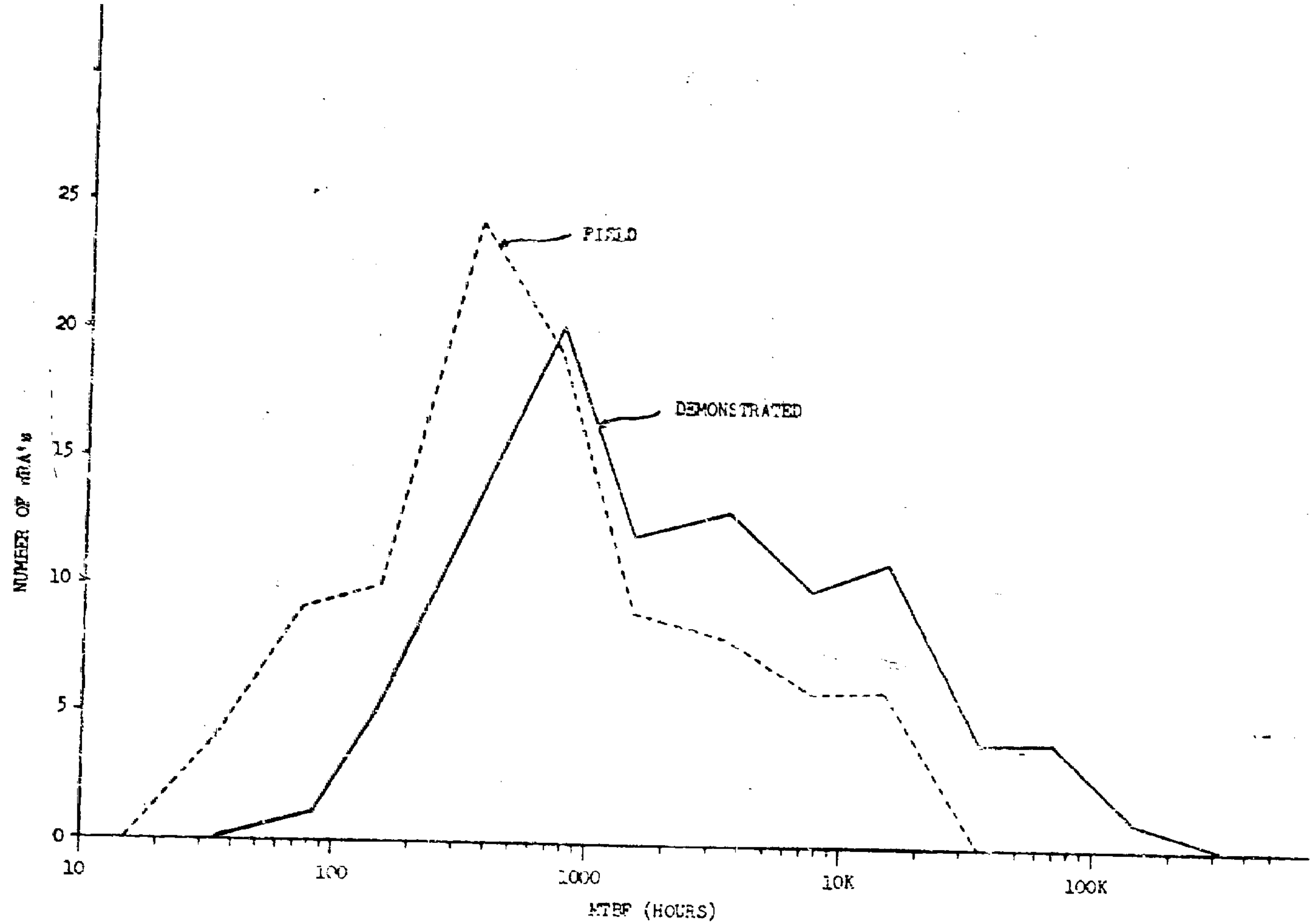


FIGURE 18 DISTRIBUTION OF DEMONSTRATED AND FIELD MTBF'S AFTER RECLASSIFICATION

do they necessarily characterize the distribution of individual differences taken over all the study items. This comparison does show in a "macro" sense that when field and demonstration experience is viewed through a consistent set of failure ground rules and common time base a somewhat more accurate picture of the differences emerges.

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Section V

STATISTICAL ANALYSIS

5.1 Quantification of Reliability Differences

The previous sections of this report described the data analyses and collection activities which were necessary to determine the reliability, environments, and conditions of use of each WRA studied during both the demonstration test and then in the field. This section describes the results of the investigation to merge this data to determine those factors that contribute to the difference between demonstrated (laboratory) and field reliability. This generally was accomplished by comparing WRA MTBF differences with differences in each factor examined to determine if any general pattern emerged.

It was first necessary to establish the measure to be used to describe the difference between demonstrated and field reliability for each WRA. It was concluded that the ratio between reclassified demonstrated and field MTBF was the measure that best suited the aims of the study. This expression was selected since it eliminated the effects of differences in failure ground rules and operating time between the laboratory and the field. In addition, large number variations and potential biases caused by using the algebraic difference between the two values were eliminated. To illustrate this point, an algebraic difference of 500 hours is a more significant difference on a 1000 hour WRA than it is on a 10,000 hour WRA. Thus, the use of relative differences provided a dimensionless scale of normalized values that can be readily analyzed. Table 11 presents the laboratory (demonstrated) and field values of MTBF, and the ratio between the two values, for each of the WRA's. A review of this table indicated that two WRA's, numbers 57 and 64, had ratios in excess of 100. No reason could be found from a preliminary review of all the data to explain the reason for these large values. It was concluded that these data points, for some undetermined reason, were so atypical that they should be eliminated to prevent these large magnitudes from seriously impacting the analysis. The

TABLE 11 COMPARISON OF FIELD AND DEMONSTRATED MTBF VALUES

WRA NUMBER	DEMONSTRATED MTBF AFTER RECLASSIFYING	FIELD MTBF (RECLASSIFIED)	DEMO MTBF/ FIELD MTBF	WRA NUMBER	DEMONSTRATED MTBF AFTER RECLASSIFYING	FIELD MTBF (RECLASSIFIED)	DEMO MTBF/ FIELD MTBF
1	381	193	1.97	21	265	1141	0.23
2	164	60	2.73	22	15,385	646	23.82
3	148	141	1.05	23	645	292	2.21
4	361	113	3.19	24	526	253	2.08
5	1,736	404	4.30	25	2,898	2,048	1.42
6	226	128	1.77	26	546	511	1.07
7	466	296	1.57	27	334	229	1.46
8	164	31	5.29	28	618	85	7.27
9	252	163	1.55	29	187	88	2.13
10	226	87	2.60	30	1,376	780	1.76
11	1,206	2,800	0.43	31	2,203	646	3.41
12	2,611	5,386	0.48	32	1,307	2,045	0.64
13	348	331	1.05	33	80	847	0.09
14	508	954	0.53	34	10,870	2,045	5.32
15	38,462	17,344	2.22	35	3,540	591	5.99
16	38,462	12,022	3.20	36	1,047	1,653	0.63
17	743	126	5.90	37	278	595	0.47
18	38,462	17,344	2.22	38	9,434	1,467	6.43
19	612	82	7.46	39	8,696	1,833	4.74
20	2,898	63	46.00	40	10,638	917	11.60

TABLE 11 COMPARISON OF FIELD AND DEMONSTRATED MTBF VALUES (Continued)

WRA NUMBER	DEMONSTRATED MTBF AFTER RECLASSIFYING	FIELD MTBF (RECLASSIFIED)	DEMO MTBF/ FIELD MTBF	WRA NUMBER	DEMONSTRATED MTBF AFTER RECLASSIFYING	FIELD MTBF (RECLASSIFIED)	DEMO MTBF/ FIELD MTBF
41	9,346	1,364	6.85	61	718	287	2.50
42	10,417	5,902	1.76	62	12,500	401	31.17
43	9,804	4,091	2.40	63	1,200	1,653	0.73
44	1,420	30	47.33	64	62,500	572	109.27
45	591	477	1.24	65	726	347	2.09
46	168	36	4.67	66	9,091	143	63.57
47	1,520	1,636	0.93	67	910	94	9.68
48	513	926	0.55	68	552	235	2.35
49	294	636	0.46	69	877	382	2.30
50	370	89	4.16	70	988	96	10.29
51	2,453	196	12.52	71	678	667	1.02
52	985	256	3.85	72	6,993	491	14.24
53	100,000	7,333	13.64	73	729	330	2.21
54	1,426	437	3.26	74	1,159	133	8.71
55	819	454	1.80	75	832	309	2.69
56	12,195	707	17.25	76	2,331	227	10.27
57	200,000	917	218.10	77	5,586	220	25.39
58	6,289	401	15.68	78	14,085	2,045	6.89
59	1,732	45	38.49	79	3,300	129	25.58
60	7,874	616	12.78	80	390	778	0.50

TABLE 11 COMPARISON OF FIELD AND DEMONSTRATED MTBF VALUES (Continued)

WRA NUMBER	DEMONSTRATED MTBF AFTER RECLASSIFYING	FIELD MTBF (RECLASSIFIED)	DEMO MTBF/ FIELD MTBF	WRA NUMBER	DEMONSTRATED MTBF AFTER RECLASSIFYING	FIELD MTBF (RECLASSIFIED)	DEMO MTBF/ FIELD MTBF
81	2,591	8,181	0.32	89	11,236	10,579	1.06
82	1,107	796	1.39	90	12,346	341	36.21
83	4,184	291	14.38	91	50,000	10,579	4.73
84	243	571	0.43	92	71,428	1,653	43.21
85	7,463	430	17.36	93	4,673	2,950	1.58
86	3,521	5,902	0.60	94	71,428	1,653	43.21
87	16,949	10,579	1.60	95	2,457	2,045	1.20
88	16,949	7,333	2.31				

distribution of the remaining 93 ratios was determined to provide some insight into how extensive the MTBF differences were from the total population. The average value of these ratios was found to be 8.2. The cumulative distribution is shown in Figure 19 and indicates that approximately 50% of the WRA's in the study had ratios of 3.0 or less and 75% had ratios of 10.0 or less.

5.2 WRA Design and End-Use Characteristics

An investigation was conducted to determine if relationship could be established between the demonstrated to field MTBF ratio and any design or end-use application characteristic of a WRA. This was accomplished by defining categories or class intervals for each such characteristic studied and then determining the average of the MTBF ratios for the WRA's falling into that category. The rationale for this approach is that if the category under study had no relationship to MTBF relative differences, the average value would be approximately the same for each interval. Conversely, if a relationship did exist, the averages would be dissimilar and the direction of the movement of the averages would provide some insight into the nature of the relationship.

The factors investigated fell into three major subgroups:

- General Design and Use Characteristics (function, weight, volume, cooling method, etc.)
- Parts (number, density, type, quality)
- Burn-In (duration, failures)

5.2.1 General Design and Use Characteristics

The relationships between the MTBF ratios and the constituent factors in this group are presented in Table 12 below. The apparent conclusions that can be drawn and probable explanations for each are:

Function: The functions appear to cluster into three groups: Interface units and RF units having the best laboratory/field correlation, Displays and Controls WRA's or Racks and Cabinets

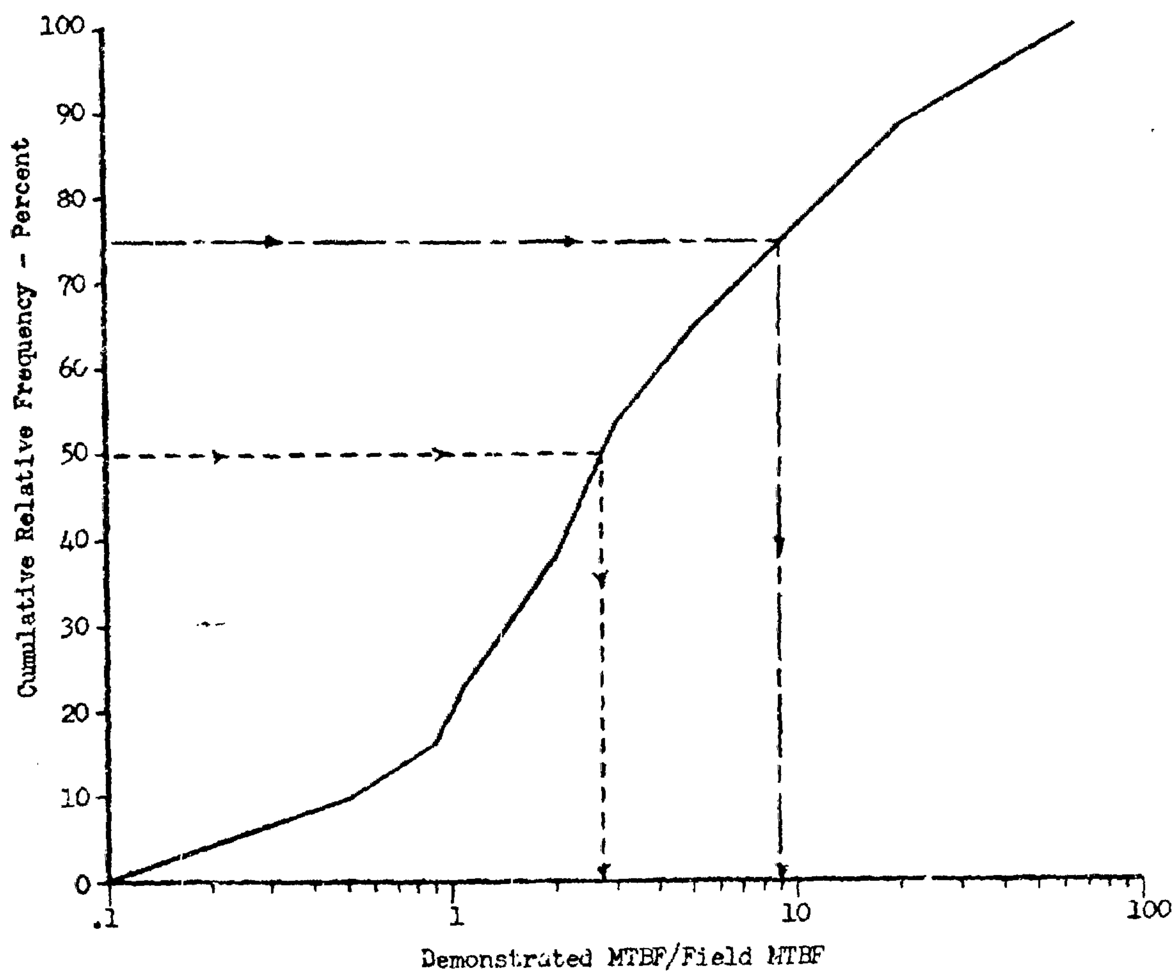


FIGURE 19 CUMULATIVE DISTRIBUTION OF DEMONSTRATION/FIELD MTBF RATIOS
— ALL WRA'S —

TABLE 12 AVERAGE MTBF RATIOS FOR SELECTED WRA DESIGN AND USE CHARACTERISTICS

Function

Function	Average MTBF Ratio*
Interface Equipment	2.5
RF Receivers and Transmitters	2.6
Computers	6.9
Power Supply	7.5
Signal Processing	7.9
Electro-Mechanical	9.2
Displays and Controls	12.7
Racks and Cabinets	22.3

Vintage

Year	Average MTBF Ratio*
Before 1970	10.9
After 1970	4.6

Weight

Weight (Pounds)	Average MTBF Ratio*
Under 20	8.8
21 to 40	8.2
41 to 80	7.7
Over 80	2.6

*Demonstrated MTBF/Field MTBF

TABLE 12 AVERAGE MTBF RATIOS FOR SELECTED WRA DESIGN AND USE CHARACTERISTICS (Continued)

Volume

Volume (Cubic Inches)	Average MTBF Ratio*
Under 100	3.2
101 to 1000	8.7
Over 1000	8.7

Power Density

Power Dissipation ($Watts/Inches^3$)	Average MTBF Ratio*
Under 0.01	11.7
0.01 to 0.10	7.5
0.10 to 0.50	8.8
Over 0.50	2.6

MIL Class

MIL Class	Average MTBF Ratio*
1A	2.4
1	6.4
2	9.4

Cooling Method

Type of Cooling	Average MTBF Ratio*
Supplemental	4.3
Ambient	11.5

Aircraft Propulsion

Type	Average MTBF Ratio*
Propeller	6.4
Jet	9.0

Mounting Method

Type of Mounting	Average MTBF Ratio*
Isolator	7.0
Hard Mounted	9.0

*Demonstrated MTBF/Field MTBF

having the worst correlation, and the remaining functions fairly closely concentrated between 7.0 and 9.0. The poor performance of the Displays and Controls could be indicative of poor duplication of the field usage conditions during the demonstration test. Usual field operations include more on/off cycling, switching of functions, crew preference adjustments, etc., than that required in the laboratory. Because of this additional usage, there is a greater tendency for equipment abuse on the part of both the flight and ground crews. Racks and Cabinets, which primarily contain connectors and interconnecting cabling, could be indicative of the differences in handling, checkout, installation techniques and differences in configuration between the demonstration test article and those that are deployed. The relatively good correlation of the RF units can possibly be attributed to more careful engineering attention to design of vehicle installation due to the criticality of EMI requirements, associated coaxial cable/waveguide runs, etc., resulting in demonstration test units more representative of end-use application.

Vintage: The WRA's of more recent design have a better correlation that those designed before 1970. This is probably a reflection of newer technology and the increased emphasis on reliability in design with the imposition of more comprehensive reliability requirements. Thus, this emphasis has resulted in more attention being paid to parts selection and application, more formal design reviews, additional testing requirements, and more thorough and systematic requirements for failure investigation and corrective action/closeout during the development and testing phases.

Weight: WRA weight does not appear to have much bearing on reliability differences except for those heavy (greater than 80 lbs.) items. Two possible explanations that may support this observation are:

- these heavy items are probably more rugged than the lighter WRA's and thus less susceptible to failure induced by the dynamic environment of the aircraft

- items this heavy generally require special handling equipment for installation and transportation from the aircraft to the shop and are thus less likely to experience any failures induced by mishandling.

Volume: Very small WRA's (under 100 cubic inches) appear to perform bettern than the larger units. These are probably very simple units with very few failure mechanisms.

Power Density: There does not appear to be any consistent relationships between dissipation per unit volume and MTBF differences. The one observation worth noting is that, contrary to prior feeling, the higher power density units show greater correlation than those with lower density. This probably results from greater design emphasis placed on assuring proper thermal environment and/or in the selection and application (e.g., derating) of components to survive in that environment. This result is consistent with the greater correlation obtained for supplementally cooled WRA's as discussed below.

MIL - Class: The more severe the end-use environment, the poorer the agreement between demonstrated and field MTBF's. This suggests that the demonstration test temperature environment less adequately duplicates the stresses of a MIL-E-5400 Class 2 temperature environment than for Class 1.

Cooling Requirement: WRA's that do not have supplemental cooling have a poorer correlation that those that do. Since supplementally cooled WRA's are more decoupled from the natural temperature environment, reliability differences are less likely to be affected by differences between the laboratory and field environments than those that are ambient cooled. In addition, ambient cooled WRA's are also more likely to experience the effects of moisture since the potentiality for water vapor migration into the WRA is greater than for supplementally cooled items.

Propulsion System: WRA's installed in jet aircraft correlate worse than those installed in propeller driven planes. This may be explained by the fact that the vibration environment during demonstration testing

(sinusoidal) more closely resembles the environment seen in propeller aircraft that than of jet aircraft (random).

Mounting Method: WRA's that are vibration isolator mounted have a small tendency to outperform those that are hard mounted. Since the purpose of special mounting is to minimize the input environment to the WRA, this observation is consistent with expectation but not as effective as one would have thought.

5.2.2 Parts

A similar investigation was performed to determine if demonstration to field MTBF differences were affected by the quantity, types, or quality of the parts used in each WRA. The measures used to describe parts characteristics were: quantity packaging density (parts/cubic in.), percent microcircuits, and percent high reliability (TX, ER or better). The values for each WRA are listed in Table E-1.

The significant relationships between these measures and MTBF relative differences are shown in Table 13 and are discussed below. No relationship was found between parts quantity and MTBF differences.

Packaging Density: No apparent relationship between packaging density and MTBF differences exists. However, it is noted that the units with the highest density (greater than 4 parts/in^3) exhibited the poorest correlation. This may be indicative of:

- field failures induced by maintenance personnel while repairing prior failures.
- localized thermal stresses occurring during field usage which the demonstration test is incapable of reproducing due to the higher thermal inertias and relatively short dwell times at temperature.

Percent Microcircuits: The better reliability correlation is associated with those WRA's that have a relatively high (greater than 50%) microcircuit population. Since, for the equipment sample of this study, virtually all of the microcircuit applications are digital, it

TABLE 13 AVERAGE MTBF RATIO FOR WRA PARTS CHARACTERISTICS

Packaging Density

Parts Density (Parts/Cubic Inch)	Average MTBF Ratio*
Under 1	8.9
1 to 2	5.1
2 to 4	7.0
Over 4	13.0

Microcircuits

% Microcircuits	Average MTBF Ratio*
0 to 50	8.9
50 to 100	2.7

High Reliability Parts

% TX, ER, Class B or Better	Average MTBF Ratio*
0 to 20	11.1
20 to 50	8.5
50 to 100	5.4

*Demonstrated MTBF/Field MTBF

follows that a higher percentage of microcircuits implies a higher proportion of digital circuitry in the equipment; consequently, since digital circuitry is relatively insensitive to temperature as well as other environmental effects, this correlation is reasonable.

Percent High Reliability Parts: As the proportion of high quality parts in a WRA increase, the correlation between the laboratory results and the field improves. This indicates that these parts are less likely to be affected by the environmental differences (primarily temperature and vibration) between the laboratory and the field than those of lower quality.

5.2.3 WRA Burn-In

The effects of WRA burn-in requirements on reliability differences were investigated. Specifically, WRA burn-in experience data for units in production was collected and several measures were defined for the analysis. The raw burn-in data for each WRA is presented in Table E-2. All the WRA's in this study were subjected to a burn-in test under environmental conditions. This was true for all production units as well as the demonstration unit. The environments were limited to temperature and vibration. The specific levels and cycle durations were generally identical with those of the WRA demonstration test.

Two measures relating to the total duration of the burn-in test were defined. One was simply the number of hours of burn-in testing a WRA experienced and the other was the number of hours divided by the predicted MTBF of the WRA. The latter measure normalized each burn-in duration to a corresponding inherent reliability capability.

The purpose for using both measures was to determine:

- if burn-in times affected the MTBF differences
- whether the absolute value or the relative length is more meaningful in specifying burn-in duration
- specific limits on the burn-in time.

The relationships with MTBF differences for each of these measures is shown in Table 14. They indicate that as burn-in time increases, the correlation between demonstration and field is better. Although some point of diminishing returns must economically exist, the data indicates that, at least until 500 hours, the more burn-in the better. The data also indicates that poor correlation exists when the burn-in duration is a very small fraction of the predicted MTBF. It further suggests that the burn-in duration should approximate or be greater than the predicted MTBF. This, of course, does not consider economic trade-offs that must be made for high reliability items. The two results taken together, suggest that, within the range of the data analyzed, a practical approach for specifying a burn-in duration requirement could encompass some variation such as:

- at least one multiple of the predicted MTBF with a minimum of 200 hours for low MTBF WRA's.
- at least 10% of the predicted MTBF with a maximum of 500 hours for high MTBF WRA's.

It should be pointed out that these conclusions are associated with a burn-in test under environments where the temperature and vibration requirements were essentially indentical with those in MIL-STD-781. Recently performed studies (Ref. 8 and Ref. 13) indicate that a more effective and shorter burn-in test (with resultant cost savings) can be realized by imposing random vibration and a more rapid thermal cycling profile as test requirements.

Data on the average number of failures per unit burned-in was also collected for each WRA. The original intent was to use this data as a gauge of the total emphasis on quality during fabrication. It was argued that since the purpose of the burn-in test was to uncover workmanship defects, the defects found should be proportional to the emphasis on manufacturing quality and inspection during fabrication. Further, since the number of burn-in failures should increase with box complexity, it was also decided to use the number of failures per part as a normalized measure of quality. The last measure defined for this analysis was the ratio of burn-in

TABLE 14 AVERAGE MTBF RATIOS FOR WRA BURN-IN MEASURES

Burn-In Hours (Production Units)

Hours	Average MTBF Ratio*
Under 100	10.9
100 to 200	5.9
200 to 500	3.9

Burn-In Hours (Production Units)/ Predicted MTBF

Interval	Average MTBF Ratio*
Under 0.001	19.3
0.001 to 0.01	8.3
0.01 to 0.10	8.4
0.10 to 1.00	5.9
Over 1.00	3.4

Burn-In Failures

Number of Failures	Average MTBF Ratio*
Under 0.5	9.5
0.5 to 1.0	7.0
1.0 to 4.0	8.0
Over 4.0	3.3

Burn-In Failures/Number of Parts

Interval	Average MTBF Ratio*
Under 0.001	7.1
0.001 to 0.01	11.3
0.01 to 0.10	24.4

Burn-In Failures/ (Hours/Predicted MTBF)

Interval	Average MTBF Ratio*
Under 0.5	10.1
0.5 to 2.0	3.1
Over 2.0	8.2

*Demonstrated MTBF/Field MTBF

failures to predicted MTBF. The rationale for using this measure was to provide an allowance for the random failure phenomena. It was argued that even if no workmanship failures were present in a WRA, random failure would still occur. In T hours of testing, the expected number of random failures would approximate T/MTBF. Thus the measure defined would describe the relative number of failures in excess of expectation.

The relationship between each of the measures discussed above and the MTBF ratios are also shown in Table 14. A review of these tables points out that burn-in failures is, in fact, an ambiguous measure. Not only does it reflect emphasis on quality, it also describes the effectiveness or efficiency of the burn-in test. Everything else being equal, the more rigorous the burn-in test, the more failures it will produce. Comparison of the relationship of number of failures with that number of failures per part illustrates this ambiguity. It indicates that the demonstration to field MTBF average ratio decreases significantly when the number of burn-in failures is greater than 4. This could be the result of such a good burn-in test that it uncovered a significant number of problems which otherwise would manifest themselves in the field. Thus the WRA would fail less often in the field resulting in a better correlation with laboratory results. Examination of the relationship of burn-in failures per part to MTBF differences shows the other side of the possible interpretation. Here, where failures are normalized to complexity, the more failures per part the worse the correlation between demonstration and field MTBF's. This illustrates the point that the degree of emphasis on quality issues during fabrication significantly impacts the correlation of laboratory and field MTBF's. These two possible interpretations, acting together, could be the explanation for the relationship of burn-in failures/expected failures to MTBF differences. For this measure, the best correlation is achieved on those WRA's that had an actual/expected burn-in failure ratio between 0.5 and 2.0. This could be interpreted to mean that poor reliability correlation between the laboratory and the field is more likely when:

- the burn-in test is ineffective and does not screen out sufficient workmanship/quality failures (ratio less than 0.5)

- the emphasis on quality during manufacture is so low that too many failures result during burn-in testing (ratio greater than 2.0)

It is recognized that any measure that uses burn-in failures directly in its derivation has limited future application. This is the only data element that is not available prior to a demonstration test since it is dependent on acceptance test data of production units which is not available until well after completion of the demonstration test. As previously indicated, burn-in failures were selected to describe emphasis on quality, with the expectation that, if it showed good correlation to MBTF differences, then it would provide the motivation and justification for a separate study, to develop an expression that quantifies the relationship among the program elements that address quality issues.

The conclusion drawn from the above analysis is that measures derived from the number of burn-in failures on production units do relate to demonstration/field MTBF differences. The proper application of these relationships, however, depends on interpretation, and it awaits some other study to determine the combined effect and an overall relationship among burn-in requirements and quality oriented parameters.

Table 15 summarizes the major conclusions drawn from the analysis of WRA design and end use application characteristics. It identifies those items which tended to produce the very strong and very weak correlation between demonstrated and field MTBF's and associtates a possible cause for each such item.

TABLE 15 WRA DESIGN AND END-USE CHARACTERISTICS SIGNIFICANTLY INFLUENCING MTBF CORRELATION

STRONGEST CORRELATION	
Factors	**Possible Cause**
WRA Function	
• RF Equipment	More careful engineering attention
More Recent Design	Improvement in reliability programs
Very Heavy	Special handling in field
Very Small	Few failure mechanisms
High Power Density	More careful engineering attention
Forced Air Cooled	More decoupled from natural temperature environment
High Microcircuit Content	Less sensitive to field environments
High "High Reliability" Part Content	Less sensitive to field environments
Effective Burn-in Tests	Screens out failures before they can occur in the field

WEAKEST CORRELATION	
Factor	**Possible Cause**
WRA Function	
• Displays and Controls	Frequent on/off cycle, field abuse
• Racks and Cabinets	Handling differences
Ambient Cooled	Closely coupled to local thermal environment, moisture
Jet Propulsion	Wrong test type
High Package Density	Maintenance induced, local thermal stresses
Poor Manufacturing Quality	Introduces additional field failures

5.3 Environmental Analysis

5.3.1 Approach

As indicated in Section I, many reasons have been advanced to explain the discrepancies between demonstrated and field values of reliability. Although the focus of this study has been on the environment, this does not suggest that it is the only factor or even the dominating factor in accounting for these reliability differences. Each of the other potential influences identified in paragraph 5.2 can and often are the reason for differences on a specific WRA. The fact that the environment alone cannot be the sole explanation is illustrated by the results of the Grumman study performed for WPAFB (ref. 1). In that study a sample of 31 WRA's were selected for analysis. These 31 items were specifically selected because the data reviewed indicated that they had experienced field failures that were environmentally induced. Even in this biased sample only 50% of the total field failures on these WRA's could be associated with environmentally related considerations. It is then reasonable to expect that in this study, where the WRA selection was much more random with respect to environmental influences, that something less than half of the influence on reliability differences is directly attributable to environmental problems.

These arguments suggest that it would be futile to expect an even near perfect relationship between an environmental factor and reliability differences. In addition, any attempt to determine a precise relationship that considers all factors, environmental and non-environmental, simultaneously would require a sample size and associated data collection effort that would be economically prohibitive.

Thus, the general approach adopted for this study was to search for and identify general trends in the data. Because of all the possible influences on any data point, the most rational approach was to analyze the data with a view to identifying, understanding, and explaining any general type of relationship between an environmental factor and reliability differences that emerges from the analysis of the data. The direction of the analysis was to identify those general trends in the data which would provide

indications and direction as to which environmental test parameters (levels, durations) were the apparent drivers and how these parameters should be modified in a Reliability Demonstration Test so as to make the results of such a test a better barometer of the reliability of the WRA that is subsequently achieved once it is in the field.

As indicated previously, the method of WRA cooling and the type of propulsion system significantly defines the environment of a WRA. It was thus decided to partition the data first by cooling method (ambient cooled and forced air cooled) and then by propulsion system (jet and propeller) to assure consistent environmental comparisons. Figure 20 shows the comparison of cumulative distributions for the two cooling methods. The comparison for propulsion types is shown in Figure 21. A review of these figures indicated that for:

- Forced Air Cooled WRA's
  - 50% of the WRA's in this group had ratios of 2.0 or less, and
  - 75% had ratios of 5.0 or less.
- Ambient Cooled WRA's
  - 50% of the WRA's in this group had ratios of 4.0 or less, and
  - 75% had ratios of 15.0 or less
- WRA's Installed on Propeller Aircraft
  - 50% of the WRA's had ratios of 1.7 or less, and
  - 75% had ratios of 5.4 or less
- WRA's Installed on Jet Aircraft
  - 50% of the WRA's had ratios 3.5 or less, and
  - 75% had ratios of 10.1 or less

again showing the influence of cooling method and propulsion type on reliability difference.

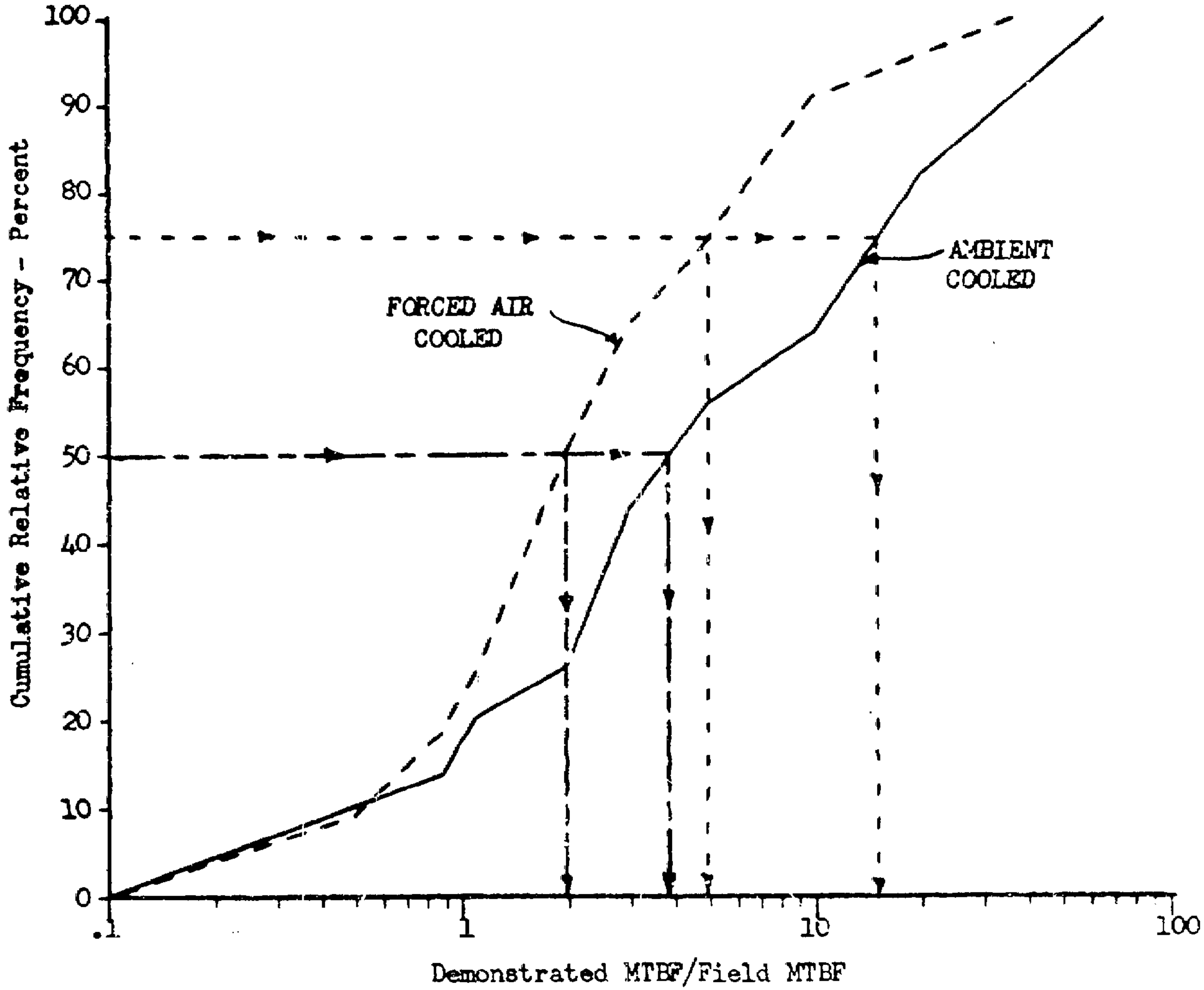


FIGURE 20 CUMULATIVE DISTRIBUTION OF DEMONSTRATION/FIELD MTBF RATIOS
— AMBIENT COOLED WRA'S VS. FORCED AIR COOLED WRA'S —

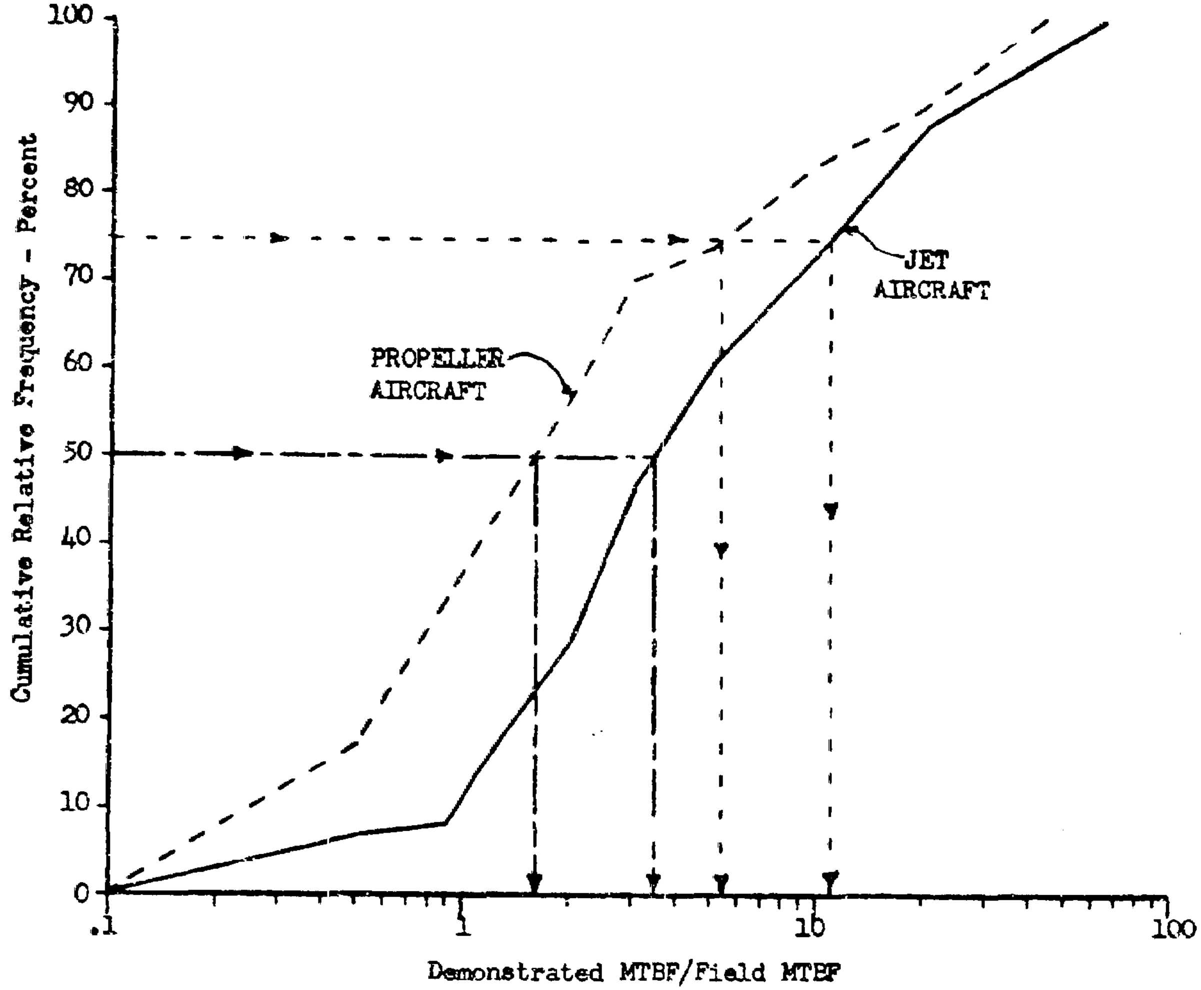


FIGURE 21 CUMULATIVE DISTRIBUTION OF DEMONSTRATION/FIELD MTBF RATIOS
— PROPELLER VS. JET AIRCRAFT INSTALLATION —

5.3.2 Environmental Parameter Selection

The major intent of this analysis was to determine which, and to what extent, differences in reliability were associated with differences between laboratory and field environments. The measure of an environmental parameter difference was taken as the ratio of the field value to the laboratory value. Here again the purpose was to establish a dimensionless measure that was independent of absolute values of the parameter. These measures may then in a broad sense be considered as the analogs to "stress ratios."

Based on a review of the environmental data presented in Appendices B and C, specific environmental parameters were selected as the basis for comparison with reliability differences. The parameters selected were those that had:

- a wide range of values in the field
- a significant range of differences between laboratory and field
- could be specified in a laboratory test

The constituent field and laboratory elements of each environmental measure used are defined in Table 16 below. In all cases the measure was evaluated as the ratio of the field value to the laboratory value. In those cases where division by zero could occur a small increment was added to numerator and denominator for all WRA's.

5.3.3 Environmental Relationships

Each of these environmental differences was investigated in turn, to determine if and how the MTBF differences changed as the value of the parameter changed. Specifically, for each environmental parameter studied, subgroups were defined, the WRA's falling into each subgroup identified, and the average of the MTBF ratios for those WRA's determined. In most cases the class intervals defined for an environment were not of equal widths. This was necessary because the data was not uniformly distributed over the entire range of values but tended to cluster at certain values. Thus the class intervals were constructed to represent the preponderance of data points and also eliminate empty class intervals. Since WRA cooling method and type of propulsion system are significant influences, it was decided

TABLE 16 FIELD AND LABORATORY COMPONENTS OF ENVIRONMENTAL MEASURES

Environmental Measure	Field Environment	Laboratory Environment
Temperature		
Low Ambient	Minimum In-Flight Compartment Temperature (°K)	Minimum Chamber Temperature (°K)
High Ambient	Maximum In-Flight Compartment Temperature (°K)	Maximum Chamber Temperature (°K)
Ambient Rate of Change	Maximum In-Flight Compartment Temperature Rate of Change (°/minute)	Maximum Chamber Temperature Rate of Change (°/minute)
Operating Time		
Low Temperature	Expected Operating Time at Low Temperature Throughout a Time Interval Equal to Total Laboratory Operating Time (Hours)	Operating Time at Low Temperature (Hours)
High Temperature	Expected Operating Time at High Temperatures Throughout a Time Interval Equal to Total Laboratory Operating Time (Hours)	Operating Time at High Temperature (Hours)
Cooling Air		
High Temperature	Maximum Temperature of Cooling Air in Flight (°K)	Maximum Temperature of Cooling Air During Test (°K)
Low Temperature	Minimum In-Flight Temperature of Cooling Air (°K)	Minimum Temperature of Cooling Air During Test (°K)
Maximum Flow Rate	Maximum In-Flight Cooling Air Flow Rate (lbs/min)	Maximum Cooling Air Flow Rate During Test (lbs/min)
Pressure		
Lowest Pressure	Minimum Atmospheric Pressure in Flight (psia)	Laboratory Ambient Conditions (14.7 psia)
Vibration		
Level (Propeller Aircraft)	Maximum Measured In-Flight (g peak)	Vibration Level During Test (g peak)
Level (Jet Aircraft)	Maximum Measured In-Flight Level (PSD)	---
Duration	Expected Flight Hours in an Interval Equal to the Total Laboratory Operating Time (Hours)	Accumulated Vibration Test Time

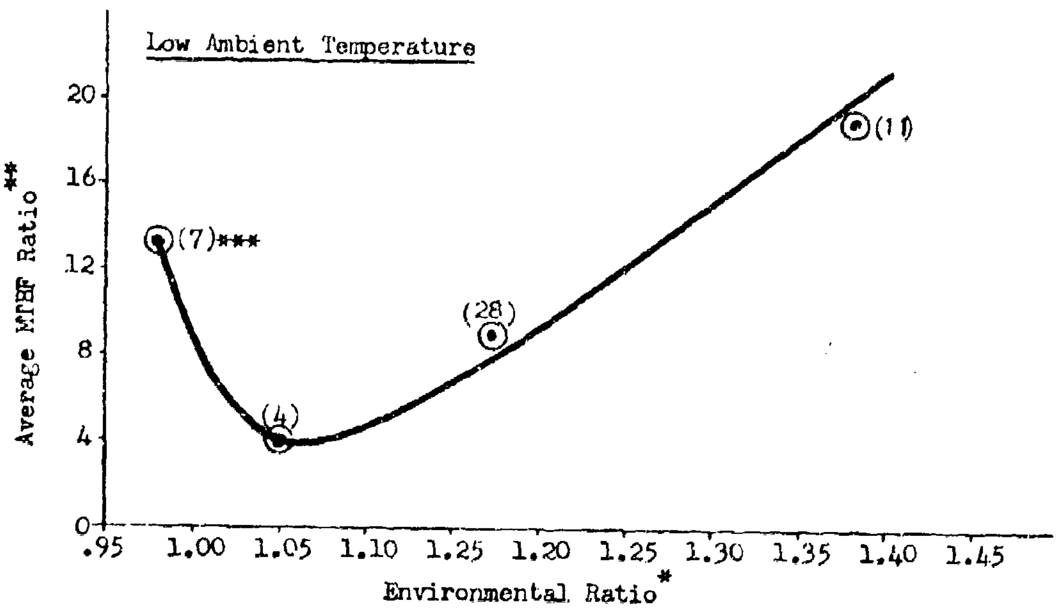
to partition the data on cooling method for the investigation of temperature related variables and on propulsion system for the investigation of vibration related variables. The relationships between these environmental parameters and MTBF differences are shown graphically in Figures 22, 23, 24, and 25. Figures 22 and 23 show the significant thermal relationships for ambiently cooled and forced air cooled WRA's respectively. The relationships involving vibration parameters are shown in Figure 24 for WRA's installed in jet aircraft while those in propeller driven aircraft are shown in Figure 25.

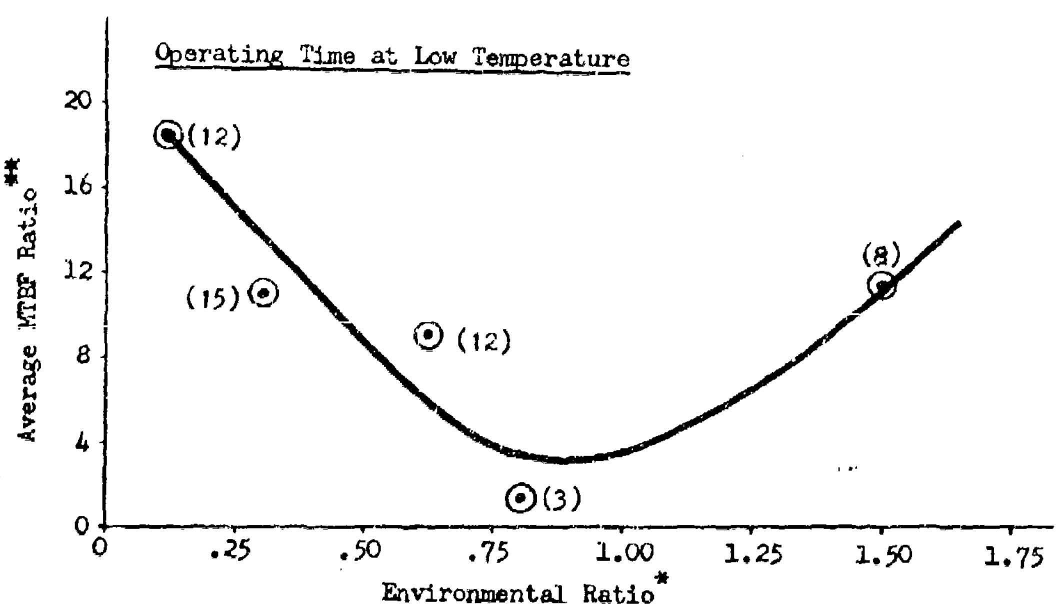
5.3.3.1 Ambient Cooled WRA's

A review of the relationships presented in Figure 22 indicates the following:

- Closer correlation between demonstrated and field reliability was achieved on those WRA's whose chamber low temperature test level approximated the inflight low compartment temperature. The average MTBF ratio of approximately 4.0 in the low temperature interval of 1.01 - 1.10 compared with average MTBF ratio of 11.5 for all ambient cooled WRA's indicates that this parameter could be a significant driver for improving laboratory to field correlation. It further suggests that the current low temperature test requirement in MIL-STD-781 of just soaking the test article at -54°C may not be an adequate low temperature test.

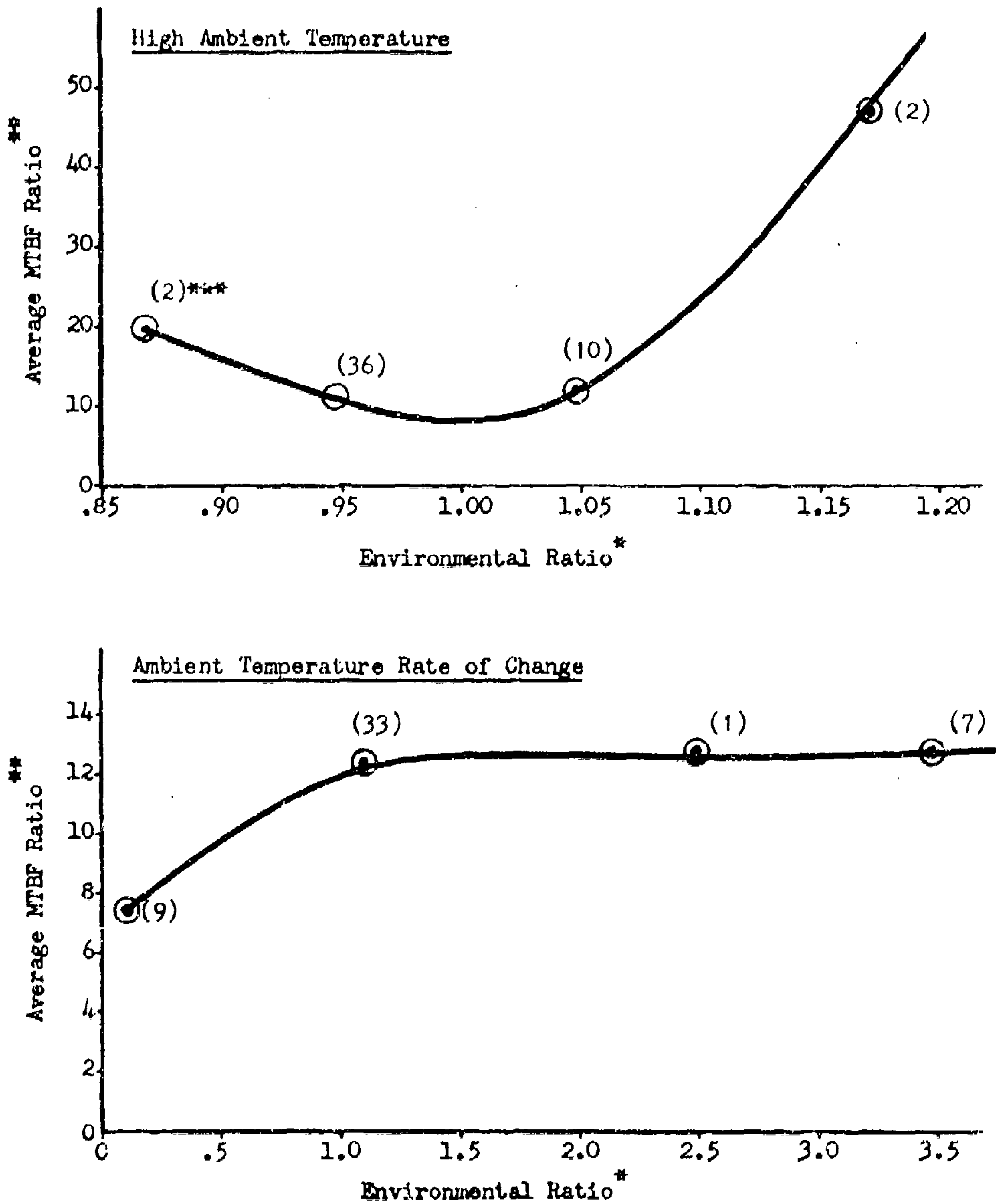
- As with low temperature, the better MTBF correlation occurred on those WRA's where the laboratory high temperature extreme closely approximated compartment high temperature conditions while in flight. This suggests that just soaking at a high temperature limit for a long period of time may not be a sufficient demonstration test requirement. Although the MTBF ratio at the low point in the curve is higher than the corresponding point on the low temperature curve, the general shape still indicates a sizeable reduction at a ratio approximating 1.0.





*Field Environmental Parameter/Laboratory Environmental Parameter
**Demonstrated MTBF/Field MTBF
***Denotes the number of WRA's represented by the data point.

FIGURE 22a SIGNIFICANT RELATIONSHIPS BETWEEN ENVIRONMENTAL AND RELIABILITY RATIOS FOR AMBIENT COOLED WRA's



*Field Environmental Parameter/Laboratory Environmental Parameter
**Demonstrated MTBF/Field MTBF
***Denotes the number of WRA's represented by the data point.

FIGURE 22b SIGNIFICANT RELATIONSHIPS BETWEEN ENVIRONMENTAL AND RELIABILITY RATIOS FOR AMBIENT COOLED WRA's (Continued)

- When ambient temperature rate of change was at least as great in the laboratory as that experienced in flight the field reliability was closer to the demonstrated value. This suggests that the $5^{\circ}C$/minute chamber temperature rate of change should probably be considered as an absolute minimum value and that requirements should be tailored to the anticipated rates expected in the field.

- Closer demonstration to field MTBF agreement occurred on those WRA's whose total operating time at low temperatures during the test more closely resembled the expected operating time at low temperature in the field. The existing MIL-STD-781 method requires that the equipment be off during the low temperature soak and be turned on when starting to raise the chamber temperature, thus never simulating in-flight operating conditions in a cold environment. This data appears to suggest that a provision to operate the equipment at anticipated mission low temperatures be required. The accumulated operating time at low temperature should be somewhat in excess (but not grossly) of that expected in the field.

- No relationship could be found for operating time at high temperature or for the low pressure comparison. This suggests that:
  - the current MIL-STD-781 high temperature operating schedule is adequate or, since so much operating time was accrued in the laboratory at high temperature, differences between that and the field are negligible in terms of their effect on reliability differences
  - pressure differences do not significantly contribute to reliability differences

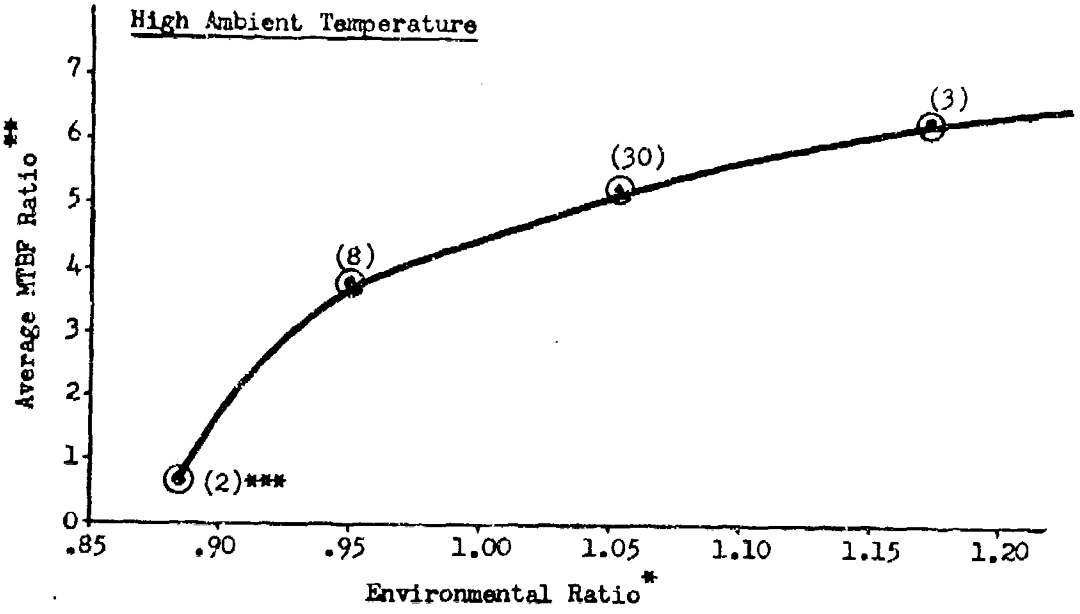
- When viewed all together, these observations suggest that for ambient cooled WRA's, the demonstration test thermal environment for at least part of the test should more closely approximate the expected in-flight environment. The combination of more representative temperature extremes and faster rates of change implies when viewed relative to the current method of demonstration testing (long dwells at extremes and moderate rates of change from one extreme to the other), that better reliability correlation would

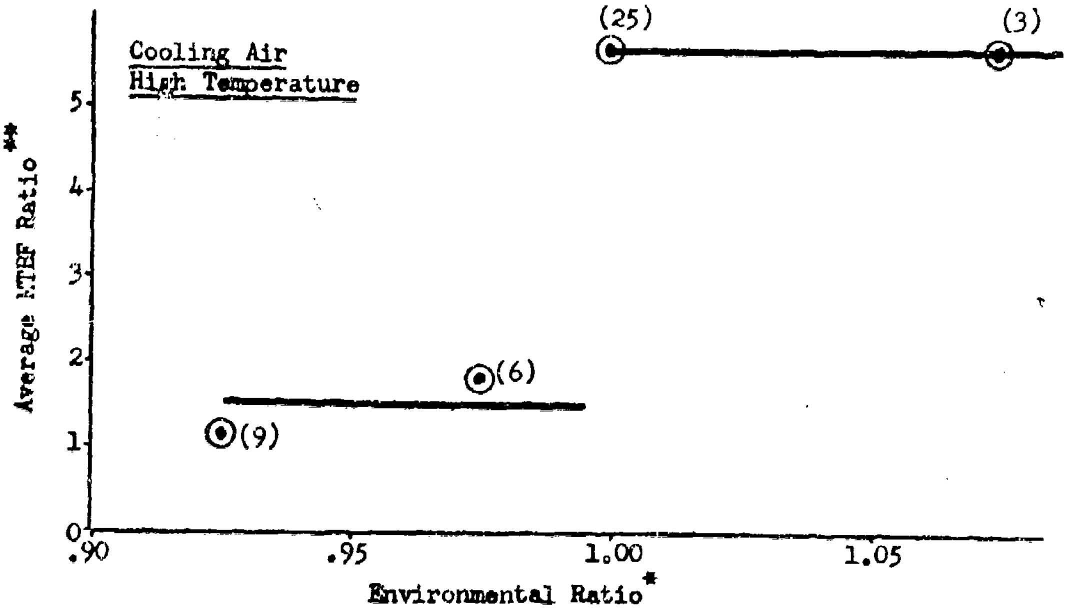
be achieved by more frequent cycling at a higher rate of change between limits. This test concept would better reproduce the thermally induced stress reversals currently experienced during field operation.

5.3.3.2 Forced Air Cooled WRA's

A review of the relationships between environmental parameters and the MTBF ratios presented in Figure 23 indicate:

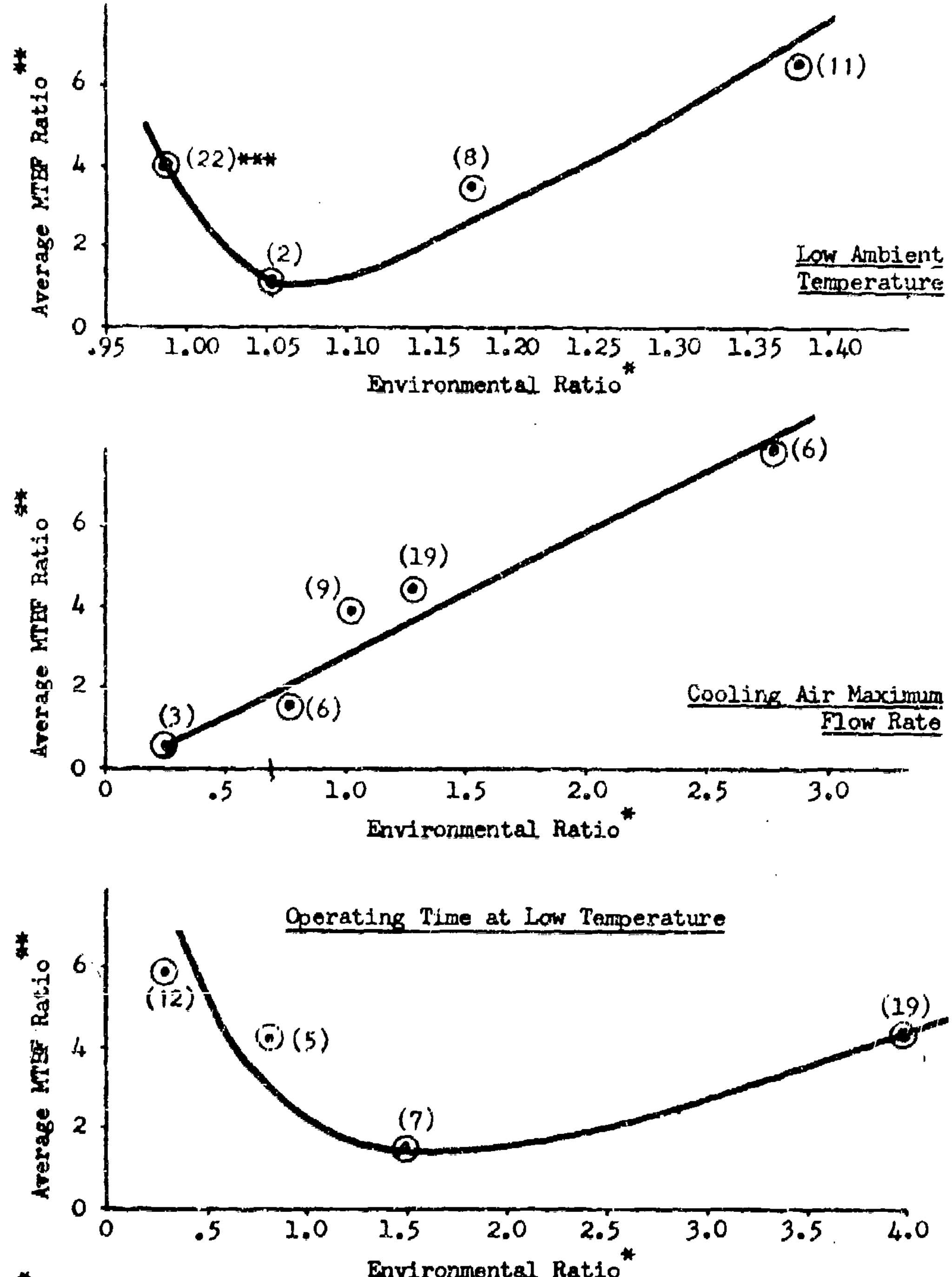
- The poorer demonstration to field MTBF ratios are associated with those WRA's where the chamber high temperature was less than the inflight compartment high temperature. In addition, the data indicates that when the maximum temperature of the cooling air was higher in the lab than in the field the reliability correlation improved. Since the requirement for supplemental cooling air is generally dictated by the amount of heat to be dissipated, these observations suggest that the high temperature environment provided by present demonstration tests is generally less severe than that experienced in the field. This indicates that the demonstration test should be structured to require, for at least part of the test, that the highest chamber temperature coincide with minimum cooling capacity consistent with the cooling limits of the WRA specification.
- The closer reliability agreement between the laboratory and the field occurred on those WRA's whose low temperature limit during the test approximated the inflight compartment low temperature. Furthermore, those WRA's subjected to a maximum cooling air flow rate in the laboratory exceeding that which it experienced in the field had the better reliability correlation. This suggests that low temperature ambient level, of itself, may not be a dominant influence. However, the current laboratory requirement of having the equipment non-operative and the cooling air off when decreasing temperature and during the soak at low temperature ($-54^{\circ}C$), may not be reproducing significant field effects. Since forced air cooled boxes are poorly coupled with the ambient temperature environment, the effect of driving the chamber temperature down without an assist from the





*Field Environmental Parameter/Laboratory Environmental Parameter
**Demonstrated MTBF/Field MTBF
***Denotes the number of WRA's represented by the data point.

FIGURE 23a SIGNIFICANT RELATIONSHIPS BETWEEN ENVIRONMENTAL AND RELIABILITY RATIOS FOR FORCED AIR COOLED WRA's



*Field Environmental Parameter/Laboratory Environmental Parameter

** Demonstrated MTBF/Field MTBF

***Denotes the number of WRA's represented by the data point.

FIGURE 23b SIGNIFICANT RELATIONSHIPS BETWEEN ENVIRONMENTAL AND RELIABILITY RATIOS FOR FORCED AIR COOLED WRA's (Continued)

cooling air is to produce a slow reduction in internal WRA temperatures (e.g., component local temperatures). Furthermore, as indicated previously, the primary thermal environment is provided by the cooling air. Consequently, if the chamber temperature was not as low, and if the soak periods were shorter than those required in MIL-STD-781, the WRA would be required to be powered on and operated more frequently. This, in turn, would require the introduction of cooling air more frequently, thus subjecting the WRA to its normal thermal environment more frequently. Since the flow rate in the field varies with mission operational parameters and since, as this data indicates, the greater the flow rate in the laboratory relative to the field the better the correlation, then a test that subjects the WRA to periodic high flow rates would be more representative of field conditions. A test with these features would better reproduce the thermal gradients and thermal stresses caused by temperature reversals.

This contention is supported by the relationship with operating time at low temperature. The data indicates that better correlation was achieved when the accrued operating time at low temperature approximated total expected operating time in the field. Since equipment operating time in the laboratory is measured from the end of the low temperature soak (when chamber temperature begins to increase), and since cooling air parameters are essentially constant, the high MTBF ratios at the extremes are indicative of insufficient variation in cooling air parameters during the demonstration test.

- No relation was found between MTBF ratios and
  - ambient temperature rate of change
  - operating time at high temperature
  - cooling air low temperature
  - pressure

indicating that either, these variables do not affect demonstration to field MTBF comparisons or, they are adequately provided for in the laboratory.

- The above observations, taken collectively, suggest that a demonstration test on forced air cooled items would result in closer MTBF agreement if it were structured to provide:
  - more frequent variation in cooling air parameters (temperature and flow rate) consistent with the item's specification limits.
  - these variations should be coupled with changes in chamber temperature
    - o on the high temperature portion of the profile so that the test article will be subjected to simultaneous high ambient temperatures and reduced cooling air capability
    - o on the low temperature side to assure more rapid and more positive cooling of components

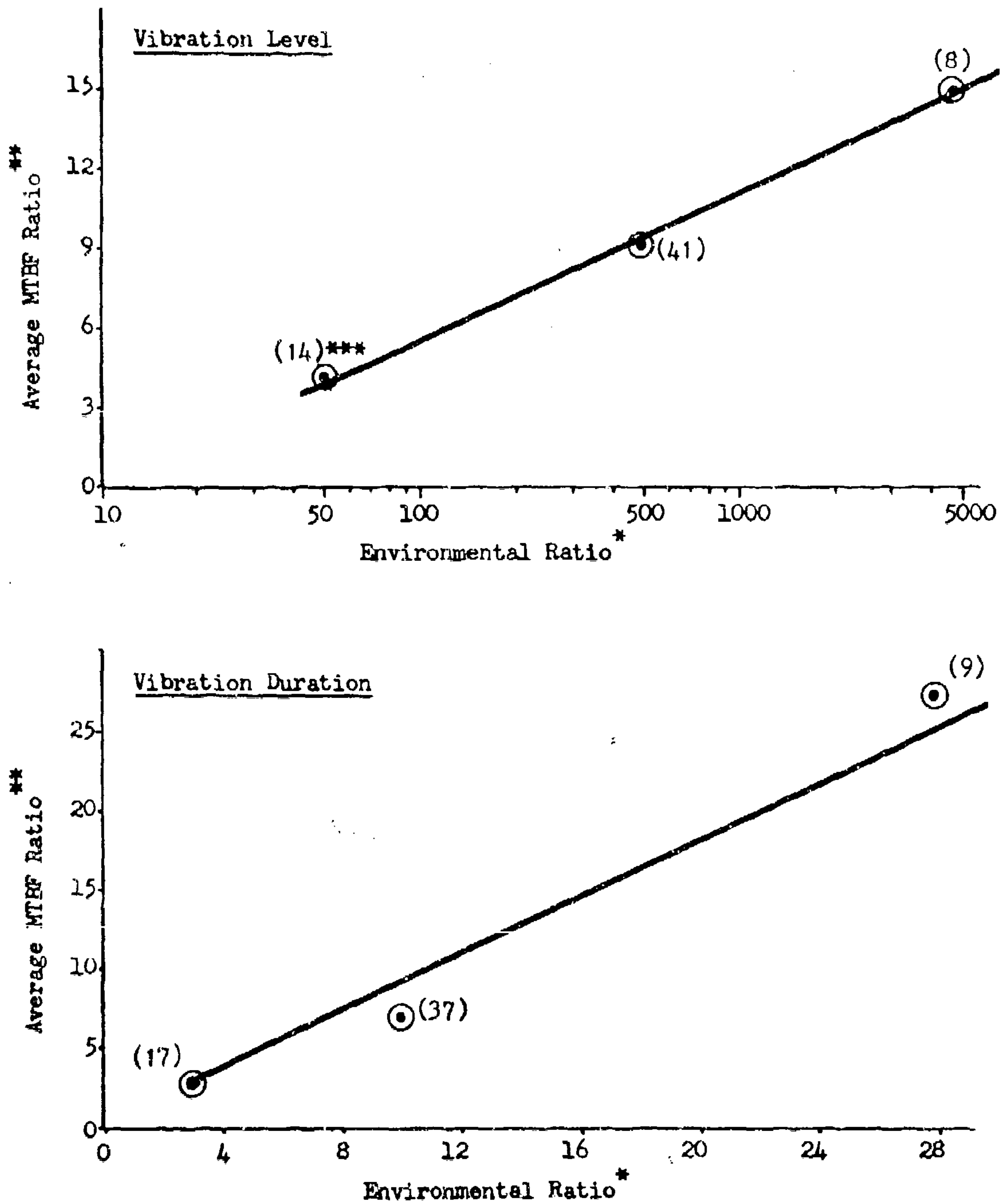
5.3.3.3 <u>WRA's Installed in Jet Aircraft</u>

The relationships between vibration parameters and MTBF ratios are presented in Figure 24 and indicate:

- The greater the field vibration level the worse the correlation between demonstrated and field MTBF's. All these units were laboratory vibration sine tested at a nonresonant frequency between 20 and 60 Hz yet the measured maximum field levels always occurred at frequencies in excess of 100 Hz. Thus the units were not tested at the higher frequencies and were subjected to an effective test level of zero PSD at these higher frequencies. Therefore, the comparison is between the maximum level in the field and zero in the laboratory. To avoid the difficulty of "division by zero" the measure ultimately used was calculated by:

  vibration measure = (Field Maximum PSD + .0001) ÷ .0001

  Thus the conclusion to be drawn from this data is that the vibration test for demonstration should more closely approximate the field environment in type, level and frequency content.



*Field Environmental Parameter/Laboratory Environmental Parameter

**Demonstrated MTBF/Field MTBF

***Denotes the number of WRA's represented by the data point.

FIGURE 24 SIGNIFICANT RELATIONSHIPS BETWEEN ENVIRONMENTAL AND RELIABILITY RATIOS FOR WRA's INSTALLED IN JET AIRCRAFT

- The better reliability correlation was assoiated with those units whose accumulated vibration test time more closely approximated the expected exposure during an equivalent operating time. The comparison of laboratory test time of 2.2g peak sine input at a non-resonant frequency vs. field time at varying frequencies and levels may be questionable, yet this very lack of agreement in test method may be magnifying the extremely high MTBF ratios at the relatively long field times.

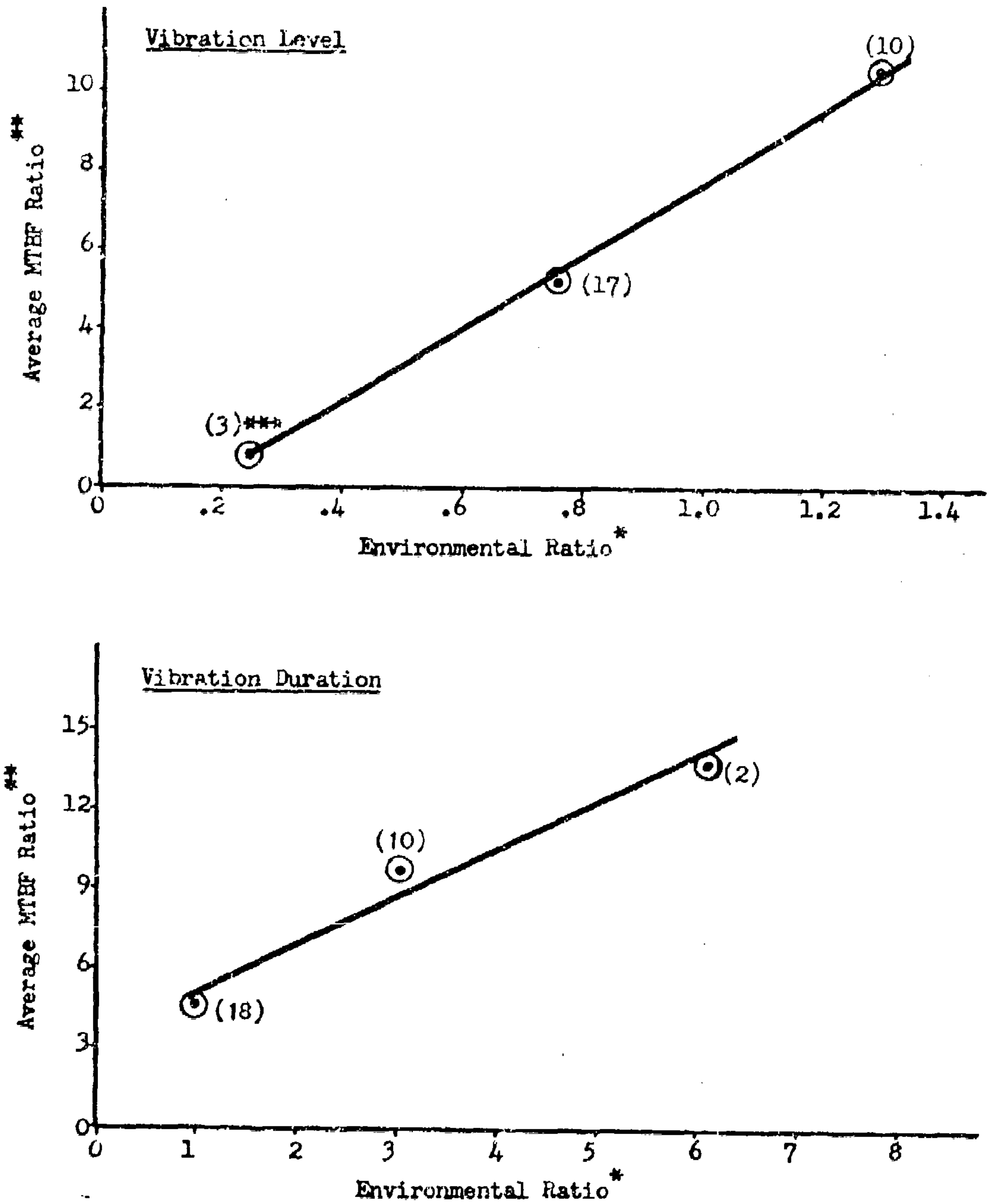
- These observations taken together suggest that a proper vibration test for demonstration purposes, on WRA's installed in jet aircraft, should be to subject the test article to a variety of input frequencies simultaneously and at levels/durations approximating field usage. This can best be accomplished by requiring that the vibration test be random instead of sine and that duration as a minimum be determined so that the relationship:

$$\frac{\text{Laboratory Vibration Time}}{\text{Laboratory Article Operating Time}} = \frac{\text{Field Flight Time}}{\text{Field Operating Time}}$$

  is preserved.

5.3.3.4 WRA's Installed in Propeller Driven Aircraft

As indicated in Figure 25, the poorer reliability correlation occurred on those WRA's whose vibration level in the field exceeded the level in the laboratory. The test and the field vibration environments were both sinusoidal in nature. However, the maximum field levels occurred at higher frequencies than those tested in the laboratory. This points out the necessity for subjecting the test article to a variety of frequencies during the test, rather than the continuous dwell at a non-resonant frequency as currently required in MIL-STD-781. This can be accomplished readily by continuous sweeping in frequency at a rate determined to assure exposure at all frequencies. This would better assure that WRA's that experience a significant vibration level in the field would be evaluated under similar conditions in the laboratory. The data further indicated that the reliability correlation is good for those WRA's that experience a relatively



*Field Environmental Parameter/Laboratory Environmental Parameter
**Demonstrated MTBF/Field MTBF
***Denotes the number of WRA's represented by the data point.

FIGURE 25 SIGNIFICANT RELATIONSHIPS BETWEEN ENVIRONMENTAL AND RELIABILITY RATIOS FOR WRA's INSTALLED IN PROPELLER AIRCRAFT

benign vibration environment in the field, thus suggesting that the current test method was adequate for this type of item.

The agreement in reliability was worse for those WRA's whose accumulated vibration test time was a smaller fraction of the anticipated vibration exposure in the field during an equivalent number of operating hours. This indicates that longer vibration test durations than those required in MIL-STD-781 would be more representative of field conditions and would consequently result in closer reliability correlation.

The above analyses indicate that closer correlation between demonstrated and field MTBF's can be achieved when the demonstration test environments more closely approximates those of the field. Table 17 presents a summary of recommended modifications to MIL-STD-781 which would assure this greater similarity in environmental exposure. The table is a listing of desirable test features (rather than specific parameter changes) which should be incorporated in a revision of the standard. Thus, these recommendations provide the general direction for the development of detailed test profiles.

5.4 Regression Analysis

Multiple regression techniques were applied to the data to establish a composite relationship between MTBF ratios and those environmental ratios appearing to have some significance. The purpose of this was to develop an expression, having the significant environmental ratios as variables, that could be used to describe the data closely. This expression then could be used to predict the consequences, in terms of effect on MTBF ratio, of alternative proposed test parameter changes.

Several different forms of a relationship were investigated. All were linear combinations of the ratios or some transformation of the ratios since anything more complex would have been difficult to interpret or apply. The expressions considered included:

TABLE 17 PROPOSED MIL-STD-781 ENVIRONMENTAL RECOMMENDATIONS

ENVIRONMENT	RECOMMENDATION
Chamber Temperatures	• Simulate compartment temperature flight levels • Cycle frequently between limits • Change levels rapidly • Provide greater assurance that components are truly exposed to indicated chamber conditions • Operate at low temperatures in accordance with expected usage
Cooling Air	• Require maximum permissible variation of cooling capability • Couple variations in cooling capability to chamber temperature variations
Vibration	• Random vibration for WRA's in jets • Sine sweep for WRA's in propeller aircraft • Levels to approximate mission levels • Increase duration to approximate flight time

$$y = a + b_1x_1 + b_2x_2 + \ldots$$

$$\ln y = a + b_1x_1 + b_2x_2 + \ldots$$

$$y = a + b_1x_1 + c_1x_1^2 + b_2x_2^2 + c_2x_2^2 \ldots$$

$$\ln y = a + b_1x_1 + c_1x_1^2 + b_2x_2 + c_2x_2^2 + \ldots$$

$$\ln y = a + b_1\ln x_1 + b_2\ln x_2 + \ldots$$

$$y = a + b_1x_1 + c_1x_1^2 + b_2x_2 + b_3\ln x_3 + \ldots$$

$$\ln y + a + b_1x_1 + c_1x_1^2 + b_2x_2 + b_3\ln x_3 + \ldots$$

where

$$y = MTBF_{laboratory}/MTBF_{field}$$

$$x_i = \text{Field Environment}_i/\text{Laboratory Environment}_i$$

Another consideration in the development of a model was the number of variables included in the expression. As a general rule, an attempt was made to keep the number of terms in the expression and, consequently, the number of variables considered, as small as possible. One can usually get apparently good fits when the number of variables is large relative to the number of data points. However, the quality of the fit can be artificial, and is analogous to perfectly fitting, as an example, a fourth degree curve through five points.

Thus, the number of variables were limited to only the previously determined significant environmental ratios, and the "fineness" of the partition of the data was limited to cooling method or propulsion type to retain at least a moderate sample size.

None of the models attempted produced a regression equation that was considered usable as a prediction tool. This again points out that other than just the environmental factors are significantly contributing to MTBF differences and even that part of the difference attributable to the environment can not be characterized simply. In several cases, however, multiple

correlation (R) coefficients of approximately 0.6 were observed. The measure, R^2, is the proportion of the variability in the dependent variable (MTBF ratios) that is explained by the function of the independent variables (the environmental ratios). Thus, 36% of the variability in the demonstrated to field reliability differences can be explained by the differences in the environments. If one assumes that variability is indicative of failure frequency, then 36% of the WRA field failures were environmentally induced that were not, or could not be, detected by the demonstration test environments. This, in itself, argues for a modification of these environments, since the potential "saving" is, on the average, approximately one third the currently experienced field failures.

SECTION VI

DEVELOPMENT OF ENVIRONMENTAL PROFILES

6.1 APPROACH

The results of the analysis of field and laboratory data, as discussed in previous sections, clearly indicates that the laboratory demonstration would be far more representative of the performance to be expected in the field, if certain reconstructions and additions were effected to the test profiles of MIL-STD-781. Recognizing that the purpose of the demonstration is to provide a measure of the expected field reliability and that the effects of natural and induced environments are one of the principal drivers affecting field reliability, it follows that the most representative test would be one that applies all of these environments at the level and in the sequence to be encountered in the field.

The revisions to MIL-STD-781 could conceivably encompass all of the natural and induced environments, however, this would be impractical in terms of effectiveness, efficiency and expense. The analysis of Section V confirms the results of previous studies indicating that environmentally induced failures are primarily attributable to temperature and vibration. Concentrated effort must be expended in these environmental areas in order to obtain better correlation between field MTBF's and demonstrated values.

The development of a cost effective, technically efficient laboratory program, applicable to a wide variety of avionics, is dependent upon certain basic ground rules as specified below:

- The specified test methods and procedures must be within the capability of standard laboratory test equipment.
- The required environmental exposure must be specified in sufficient detail to assure that the test article receives the full effect of the exposure.
- The method of developing the specific test profile should accommodate the use of preliminary aircraft and WRA performance definitions.

- The developed program should provide a high ratio of equipment on time to test time.
- The test set-up should allow adequate performance monitoring before, during and subsequent to each environmental exposure.
- The means utilized for performance monitoring should provide positive recognition of a failure.
- The success/failure criteria should be clearly defined and consistent with the criteria to be utilized in measuring field performance.

The selected approach to developing a laboratory program which is essentially analogous to the temperature, moisture and vibration environments expected in the field, utilizes an aircraft mission as its base. Although aircraft with different mission goals have different profiles (as described by an altitude - speed time schedule) certain generalizations of profile can be made. Every aircraft experiences the following sequence of events during one nominal mission:

- Ground operation
- Take-off and Climb to Altitude
- Mission Objective
- Descent and Landing
- Ground storage (Non-operation)

Utilizing this general sequence, one can then identify performance parameters associated with the various phases within the sequence. Furthermore, since the parameters can be expressed in terms of speed, altitude and duration, a viable approach to the development of a test program which is the analog of an aircraft mission, becomes apparent.

Separating the environments of concern into their constituent parts, the method by which the aircraft mission parameters can be utilized in defining the environmental levels and durations evolves as follows.

As discussed in Section III a WRA's thermal time-history is a function of the ambient environment, duty cycle, cooling method and electrical power density characteristics. Clearly, the operational ambient environment is an aircraft/aircraft mission dependent variable. The duty cycle and

electrical power density are fixed values, specified by the design specification, and the cooling method effect, for WRA's other than ambient cooled, can be defined as a function of cooling airflow/temperature.

If one considers the laboratory thermal chamber analogous to the WRA's aircraft compartment, a chamber thermal profile can be developed based upon expected compartment temperatures during the various aircraft mission phases. Furthermore, if these expected temperatures are determined for hot and cold day extremes, the chamber profile will encompass the full compartment thermal ambient range. Coupling the chamber profile, thus developed, with the equipment's nominal duty cycle and cooling schedule will produce an exposure which is truly the thermal analog of service conditions.

The laboratory vibration environmental levels and durations can be similarly developed using those parameters associated with the vibration environment. This vibration exposure can then be combined with the previously developed thermal exposure to yield the required mission environmental analog.

High levels of moisture exist as a natural field environment and avionics are periodically subjected to these extremes. Furthermore, the results of previous studies (reference 1) conclude that moisture is one of the prime environmental drivers of avionic failures. Therefore, in order to correctly simulate the major environments, to which WRA's are normally subjected during their service life, a periodic humidity exposure must be included in the laboratory program.

The development of a practical and economic laboratory program is in part based upon the consideration of all of the constraints and limitations associated with environmental testing. Although practically the full range of steady state environmental conditions can be reproduced singularly using standard laboratory equipment, high rates of change associated with transient conditions and combined environmental exposures are far more difficult to correctly duplicate.

Dividing the "environmental world" into its mechanical and climatic constituents, one can readily see that the imposition of a mechanical environment (vibration, shock, acceleration) relies solely upon the controlled transfer of energy into the test article. Given then, that a specified

mechanical environment can be generated by a mechanism of sufficient force output, the constraining factor is the force input direction; i.e., single axis excitation. Climatic environments (temperature, pressure, etc.) however, require the ability to transfer energy both into and out of the article. If one assumes that sufficient energy can be made available for input to the article then the constraining factor to single environment climatic testing is the capability of the mechanism utilized as a sink or reservoir for the storage of removed energy. Summarizing the above discussion, relative to single environment testing, one concludes that the duplication of a mechanical environment is limited to single axis excitation (one axis at a time) and climatic environmental duplication is limited by the capability of the available energy sink.

Investigating the relationship between the laboratory generated environments and those existing within the near space of earth, one can again determine that the mechanical environments are more closely coupled to the mechanisms of the vehicle than are the climatic environments. Vibration, shock and acceleration in an aircraft are all a function of power plant, velocity, velocity changes fluid density and directional stability. Each of the resultant conditions associated with these parameters is readily duplicated in the laboratory as steady state conditions one axis/environment at a time. The climatics, however, are not solely a function of the parameters associated with the vehicle. At any instant in time, each of these climatics exists as stabilized multi environmental layers within the envelope of the earth's atmospheric expanse. The high performance aircraft, flying through these various layers, experiences rapid climatic changes due to its direction and velocity not due to any change within the stabilized climatic layers.

Reviewing the foregoing paragraphs, one recognizes that the primary difficulties associated with accurately duplicating a high performance aircraft's environmental time line within a laboratory, is:

- Attaining the high rates of change.
- Duplicating various climatic environments either simultaneously or sequentially within very short time periods.

- Recycling climatic environments to produce an analog of multiple high performance climbs and dives.
- Generating a mechanical environmental spectrum which accurately, simultaneously produces the multiple modes of excitation and rapidly varying levels.

Modern laboratory environmental generating equipment is designed around the requirements of Military Specification such as MIL-STD-810. The documents prescribe accurately achieved and maintained steady state conditions or repeatable spectra. The enumerated tests therein are not constructed to be an analog of the transients associated with actual flight. Their purpose is to provide a method of imposing "qualification" level environmental stresses upon a test article to obtain a measure of its safety margin. Since the test equipment is designed to comply with the rigid requirements of these environmental test methods, rapidly changing transient conditions may require minor alteration of the equipment.

The herein presented profiles have been developed with full cognizance of the limitations of standard laboratory test facilities. Within the constraints of economics and generally available laboratory equipment capability, the developed profiles can be achieved by incorporating the following recommendations:

- Augment the capability of a standard temperature chamber by the use of external temperature conditioning unit(s).
- Rework the temperature controller so that it controls the external unit(s) in addition to the chamber's heating and refrigeration equipment.

The effective implementation of the developed environmental profiles requires that each environmental sequence be conducted to the high level of excellence specified in the various "Environmental Test Method" Military Standards such as MIL-STD-810. In order to achieve this goal, it is recommended that the appropriate sections of MIL-STD-781, associated with this issue, be modified and/or amplified to specify pertinent parameters as outlined below.

Test Plans and Procedures

The approved test plans and procedures shall include specific definition of the test sequence, equipment, methods, safety requirements and data sheets. The document(s) shall also contain a detailed instrumentation plan which specifies the data acquisition requirements and the performance characteristics of the equipment to be utilized in fulfilling these requirements.

Test Reports

The reports shall include all supporting data collected in conducting the tests, and analysis of all failures which occurred. The organization of the test report shall correspond to that of the approved test plan and the presentation of the test report shall be responsive to the requirements of the test plan/procedure. All test data shall be signed and dated by the test engineer for certification.

General Test Requirements

All testing shall be accomplished in accordance with the applicable requirements specified in any of the approved Military Standards for Environmental Test Methods such as MIL-STD-810. All of the general requirements such as standard ambients, measurements, tolerances, accuracy of test apparatus etc., should be specified or at least referenced. The specified requirements, associated with the generation and application of each applicable environment, should either be referenced to an Environmental Test Method or specified.

6.2 TEMPERATURE AND HUMIDITY

6.2.1 Major Considerations

As previously discussed, the laboratory program should represent the environmental stresses which the WRA experiences during its service life and as such should encompass the following phenomena:

- thermal environment in flight
- thermal environment on the ground
- high and low temperature start-ups
- periods of operation and non-operation

• humidity exposure

Although the parameters associated with flight can be defined utilizing the mission analog approach previously discussed, those parameters associated with non-operating, storage periods must be represented by calculated equivalents. Recognizing that the qualification test verifies the equipment's capability to successfully survive steady state environmental exposures, the purpose of non-operating dwell periods in this program must be otherwise defined. The reasons for including these dwells are to assure equalization of chamber and equipment temperatures following each mission analog and to produce an effective humidity exposure. Combining those periodic, non-operating dwells with the mission analog approach yields a test cycle which in general terms represents real time/levels during the operating flight phases and effect-equivalent times/levels during the non-operating ground phases.

The analysis presented in Section III concludes that the initial (ground) ambient temperatures, realized during the aircraft field use, fall within temperature extremes characterized as a "cold day" and a "hot day". Since the WRA thermal profile is in part, a function of the initial ambient temperature, the laboratory program is constructed utilizing repetitive cycles alternating between cold and hot days (refer to Figure 26.)

Field data indicates that the aircraft is periodically stored at extreme moisture conditions for extended periods of time, however, MIL-STD-781 does not require an evaluation of the ability of the design to withstand these periodic exposures. Inasmuch as no test was performed in the laboratory, environmental comparisons, similar to those presented in Section V, were not performed for moisture. An investigation of the field failures on the study WRA's was performed to determine the extent of the moisture problem on these items. Field failure reports were reviewed and a significant number of those attributable to environmental causes were due to moisture related reasons. The predominant manifestations were shorting of components, corrosion, water entrapped in the unit, and salt deposits on cards. In view of this result, and the conclusions drawn in reference 1

Phase	Test Phase Definition	Duration	Operating/Non-Operating
A	Ground Operation - Cold Day	30 Minutes	Operating
B	Climb to Altitude	Defined by each Weapon System	Operating
C	Mission Objective	Defined by each Weapon System	Operating
D	Descent to Hot Day	Defined by each Weapon System	Operating
E	Ground Storage - Hot Day	30 Minutes	Non-Operating
F	Ground Operation - Hot Day	30 Minutes	Operating
G	Climb to Altitude	Defined by each Weapon System	Operating
H	Mission Objective	Defined by each Weapon System	Operating
I	Descent to Cold Day	Defined by each Weapon System	Operating/Non-Operating (*)
J	Ground Storage - Cold Day	30 Minutes	Non-Operating (*)

(*) This non-operating time is desired to be 1 hour total. This period is derived from 1/2 hour which is Phase J and 1/2 hour from the total allocated for Phase I. However, if the total Phase I time is less than 1/2 hour, the total non-operating time will be similarly reduced.

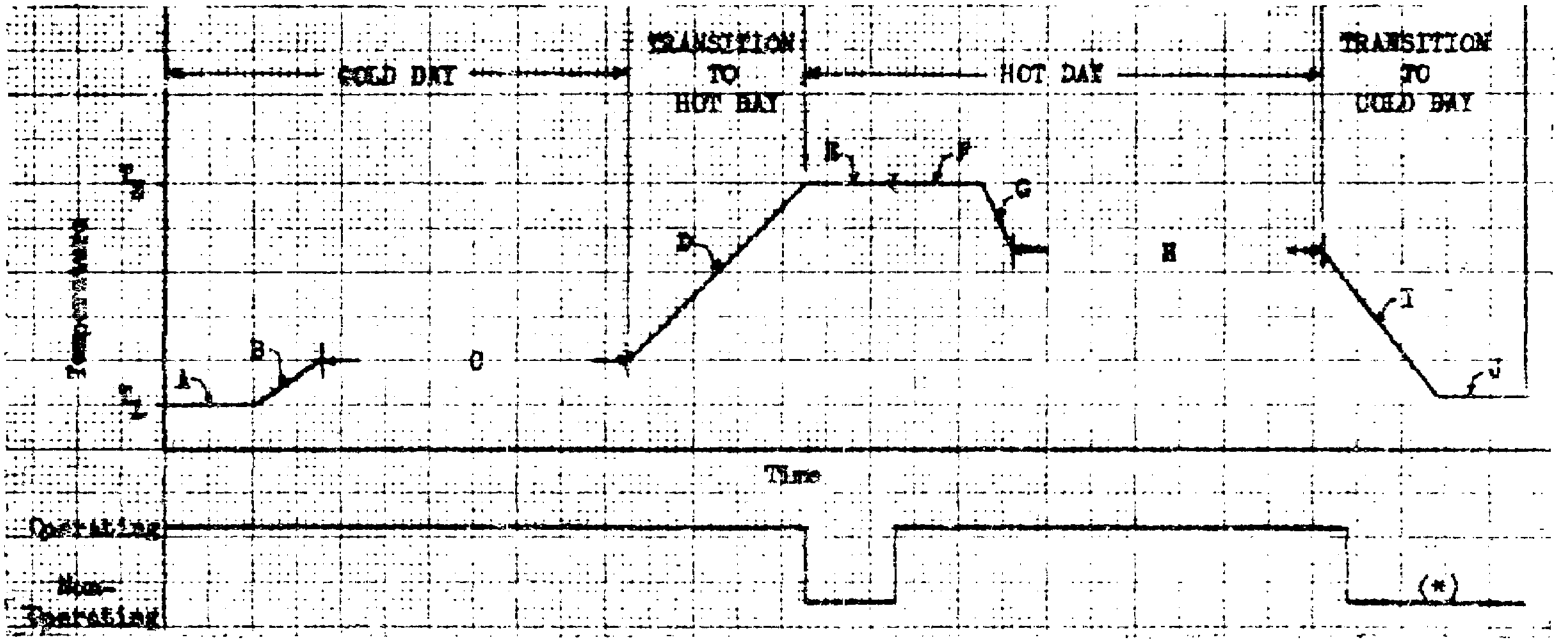


FIGURE 26 TYPICAL MISSION THERMAL CYCLE PROFILE

regarding the problems with moisture in the field, the inclusion of a moisture evaluation as part of the demonstration test is considered essential.

Ideally, any proposed humidity test should simulate the full range of moisture environment expected during service life. Realistically, this full range, which encompasses all conditions between hot day - high relative humidity ground storage and high speed climb/dive through varying thermal/pressure layers of atmosphere, cannot be practically duplicated in a laboratory. Recognizing these limitations, the standard test methods, i.e., MIL-STD-810, base their approach upon manipulating certain of the environment's driver and driven constituents to produce the desired long term life effects. The same approach has been utilized to develop a cycle for this program, however, since the reliability demonstration is an extrapolated program, designed to represent a percent of real life, the total cyclic exposure has been reduced and dispersed throughout the extent of testing (refer to figure 27.)

6.2.2 Durations and Levels

Based upon the conclusions drawn from the analysis presented in Section V, profiles recommended herein, seek to vary test conditions as they would in a true aircraft mission profile and as such, are a deviation of the mission variable WRA and compartment thermal parameters. In order to assure obtaining the full effect of applied environmental exposures, certain compromises must be made to the mission analog approach.

The effects of high rate of change thermal cycling manifest themselves as fatigue failures caused by thermally induced stress reversals. The approach to duplicating the natural pheomena in the laboratory, is to artifically manipulate those parameters which affect the thermal time history of a WRA, so that the end product is an analog of the expected service conditions.

The analytical results presented in Section V indicate that higher rates of thermal change than those currently employed in the test program may be advantageous. Although MIL-STD-781 currently requires a minimum

CYCLE NO.	*0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	
CYCLE TYPE	COLD SOAK (Fig. 28)	THERMAL**	THERMAL	THERMAL	THERMAL	HUMIDITY	THERMAL	THERMAL	THERMAL	THERMAL	THERMAL	THERMAL	THERMAL	THERMAL	THERMAL	THERMAL	THERMAL	THERMAL	THERMAL	THERMAL	THERMAL	REPEAT FROM CYCLE NO. 1

*Cycle "0" is used only at start of test and any time test article has been allowed to dwell at room ambient.

**Thermal cycle is defined by the equivalent of Fig. 26.

FIGURE 27 SEQUENCE OF 20 CYCLE SET

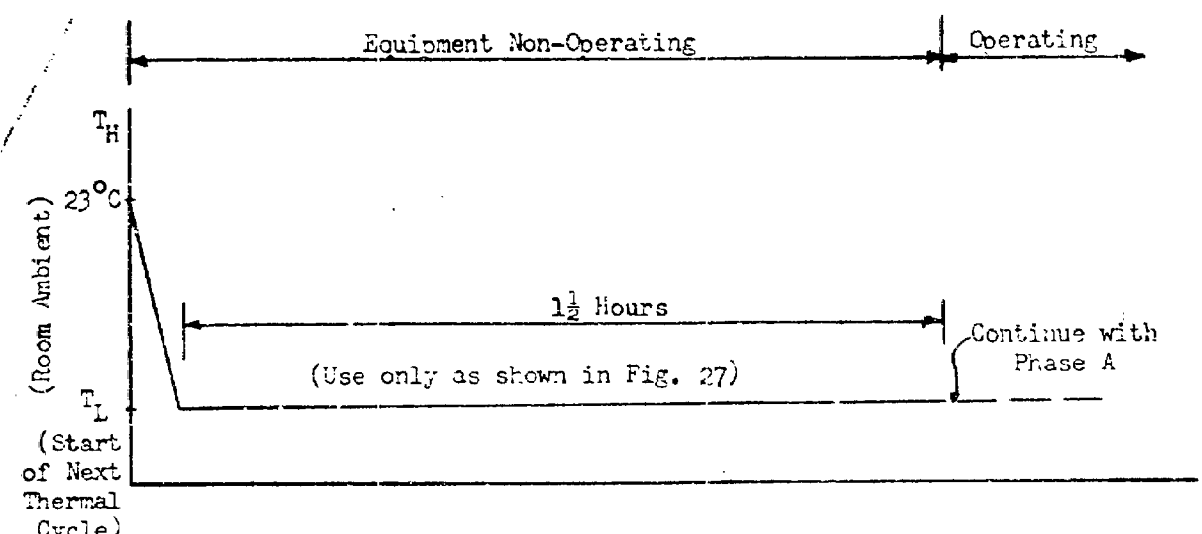


FIGURE 28 "ZERO" CYCLE CHAMBER PROFILE (COLD SOAK)

rate of 5°C/min, this value has often been used as a maximum requirement regardless of the anticipated field conditions. Based upon this conclusion, the temperature rate of change is established at the average expected field rate for each change in steady state condition. A minimum rate of change of 5°C/min is recommended for developed thermal profiles containing a lesser rate.

6.2.2.1 Test Duration

The duration of each operating flight exposure is the real time analog of the applicable aircraft mission (refer to Figure 26). The duration of operating and non-operating ground exposures is based upon average field data and laboratory experience. The non-operating periods separating each hot and cold day cycle are asymmetric in order to take advantage of the demonstrated ease in achieving avionic equipment high temperature stability. The one-half hour non-operating dwell (Phase E) is considered sufficient to achieve equalization between the chamber and equipment temperatures. In order to assure equalization at low temperatures, a nominal one hour non-operating period is specified between repetitive cycles (portion of Phase I and entire Phase J). The ground operating time is specified as one-half hour per "mission" as typical of the warm-up/check-out.

The duration of the humidity exposure is the product of an approximation of the total expected field effect and the limits dictated by test equipment operation. The total exposure, which is a derivative of MIL-STD-810 cycle, is distributed throughout the entire test duration every twenty basic thermal cycles to yield a total humidity exposure of approximately 20% of the total test time.

6.2.2.2 Test Levels

Analysis of the field environmental data collected for this study is the basis for the recommended test levels. Data provided on certain additional aircraft has been reviewed and included to supplement data on the aircraft in this study in order to define the minimum and maximum thermal levels.

Cold Day Temperature Levels

Reviewing the data for minimum temperature levels indicated a wide

variation of compartment minimum temperature. Three distinct bands were noted in the vicinity of -54°C, -18°C and +10°C, at sea level. The three bands reflect the extent of coupling between the compartment and the outside ambient temperature. Compartments closely coupled with the outside environments represent the lowest temperature band, while the highest band represents a minimum coupling. These minimum temperatures represent steady state conditions and will be achieved when the mission profile allows sufficient time to overcome the thermal inertia of the compartment. Generally, compartments located close to the aircraft skin have the greatest thermal coupling to the outside ambient, while those close to the aircraft centerline have the least coupling. However, a more precise estimate of the compartment minimum temperature is a complex function which is ordinarily evaluated during the routine thermal design of the aircraft and WRA.

The amount of applicable flight data at cold day conditions was limited. Thus the development of cold day temperature levels was based on a combination of standard atmospheric conditions and engineering judgement.

The coldest test level (-54°C) was constructed by taking the cold day definition of temperature vs. altitude between sea level and 20,000 feet. To complete the curve, a minimum temperature of -50°C was established for altitudes above 30,000 feet. This reflected the minimum compartment temperature observed under field environments and review of additional data. This curve was then displaced linearly by the appropriate amounts to arrive at the levels representing -18°C and +10°C at sea level initial conditions. The available flight data was plotted against these developed curves and is presented in Figure 29. It shows that the curves are reasonable representations of field experience and thus may be used for testing purposes. The levels are tabulated in Table 18 and are valid for Class I and Class II equipment.

Hot Day Temperature Levels

The MIL-E-5400 definition of temperature vs. altitude is the Hot Day temperature level for Class I equipment (refer to Table 19.) Observation of the field environment data indicates that a majority of the Class I

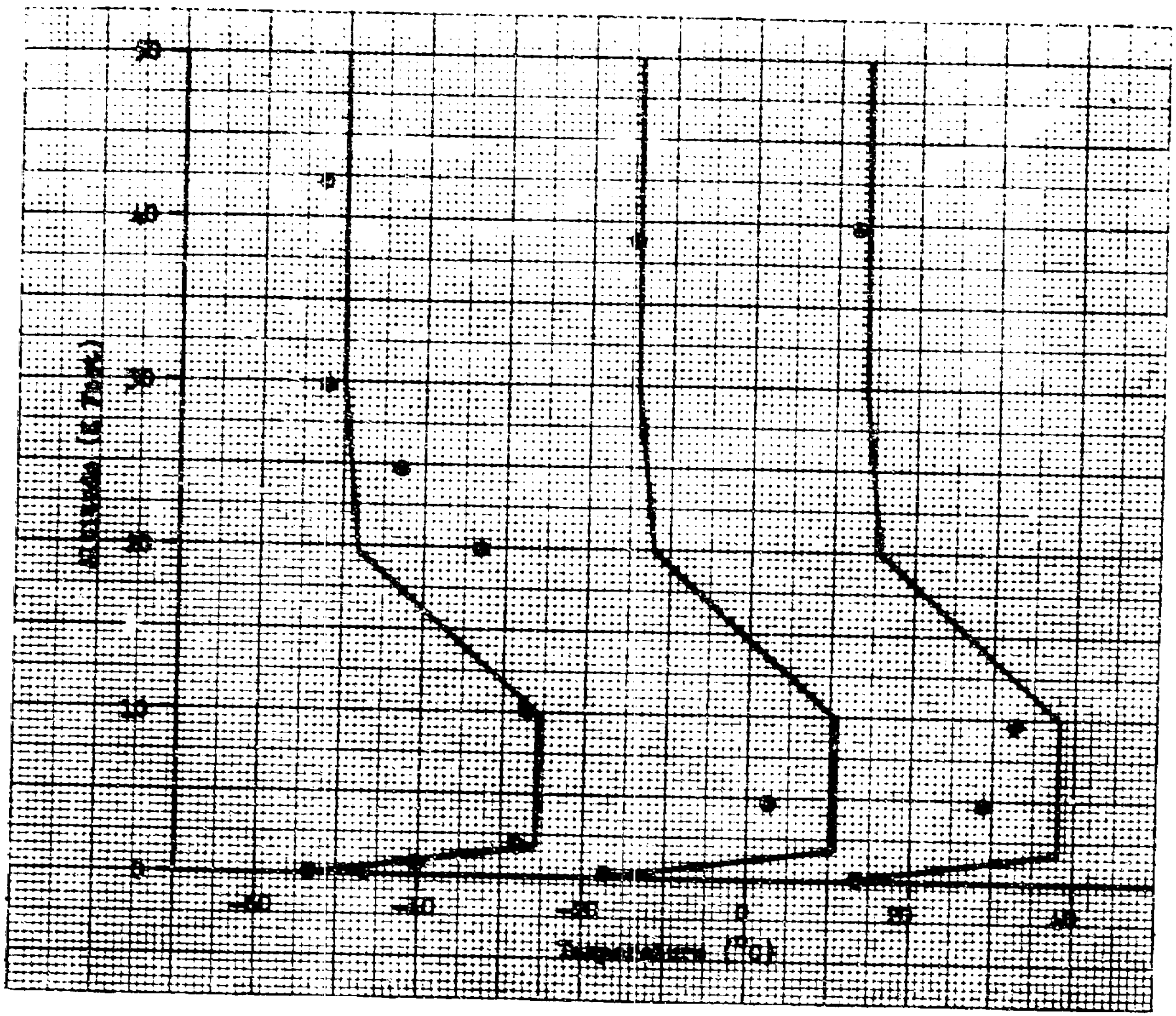


FIGURE 29 COMPARTMENT MINIMUM TEMPERATURE BANDS

TABLE 18 COLD DAY COMPARTMENT AMBIENT TEMPERATURE (^{o}C) FOR CLASS I & II EQUIPMENT

Altitude (K feet)	Temperature (Minimum Compartment, Steady State) (1)		
	$-54^{o}C$	$-18^{o}C$	$-10^{o}C$
0	-54	-18	-10
2	-26	10	38
10	-26	10	38
20	-48	-12	16
30	-50	-14	14
40	-50	-14	14
50 - 70	-50	-14	14

(1) Select the column which most closely reflects the minimum expected steady state compartment temperature.

TABLE 19 HOT DAY COMPARTMENT TEMPERATURES (^{o}C) FOR CLASS I EQUIPMENT

Altitude (K feet)	Temperature (^{o}C)
0	55
10	53
20	40
30	40
40	30
50	20

equipment analyzed has a maximum temperature of 27°C. However, this temperature represents the average of the cockpit normal flight temperature. The equipment's ambient environment would be expected to reach the MIL-E-5400 limit of 55°C due to packaging considerations, i.e., the avionics in the cockpit are closely stacked in a panel resulting in localized temperature higher than the average cockpit ambient temperature.

For Class II equipment, the compartment temperature data was plotted versus altitude and mach number as shown in Figure 30. For a mach number of 1.0 and greater, the data correlated fairly well with the continuous and intermittent operating limit of MIL-E-5400 respectively. Therefore, the MIL-E-5400 curves were used for the recommended test profiles for these two speed conditions. For lower mach numbers, curves were approximated to the data, constrained by MIL-E-5400 "Hot Day" sea level requirements and biased to the high side. For high altitude conditions, lack of field data necessitated an approximation based upon experience. The curves and resulting test levels are presented in Figure 30 and Table 20 respectively.

Cold and Hot Day Ram Cooled Compartment/Equipment Temperature Levels

The presented levels (refer to Tables 21 and 22) to be used in the construction of profiles for compartments or equipment which are ram air cooled, are derived. They are based upon:

$$T = T_{AMB} (1 + .2r M^2)$$

where:

T = Ram Temperature ~ °K

T_{AMB} = Outside Ambient ~ °K

r = Recovery Factor

M = Mach Number

The recovery factor is a measure of the action of the free-stream dynamic-temperature rise recovered at the surface. The factor was assumed to equal 0.9 which is an accepted value for a turbulent boundary layer. Table 21 presents hot day levels and Table 22 presents cold day levels.

6.2.3 Forced Air Cooling

One of the primary drivers to the internal temperature of forced air cooled WRA's is the temperature/flow characteristics of the cooling air.

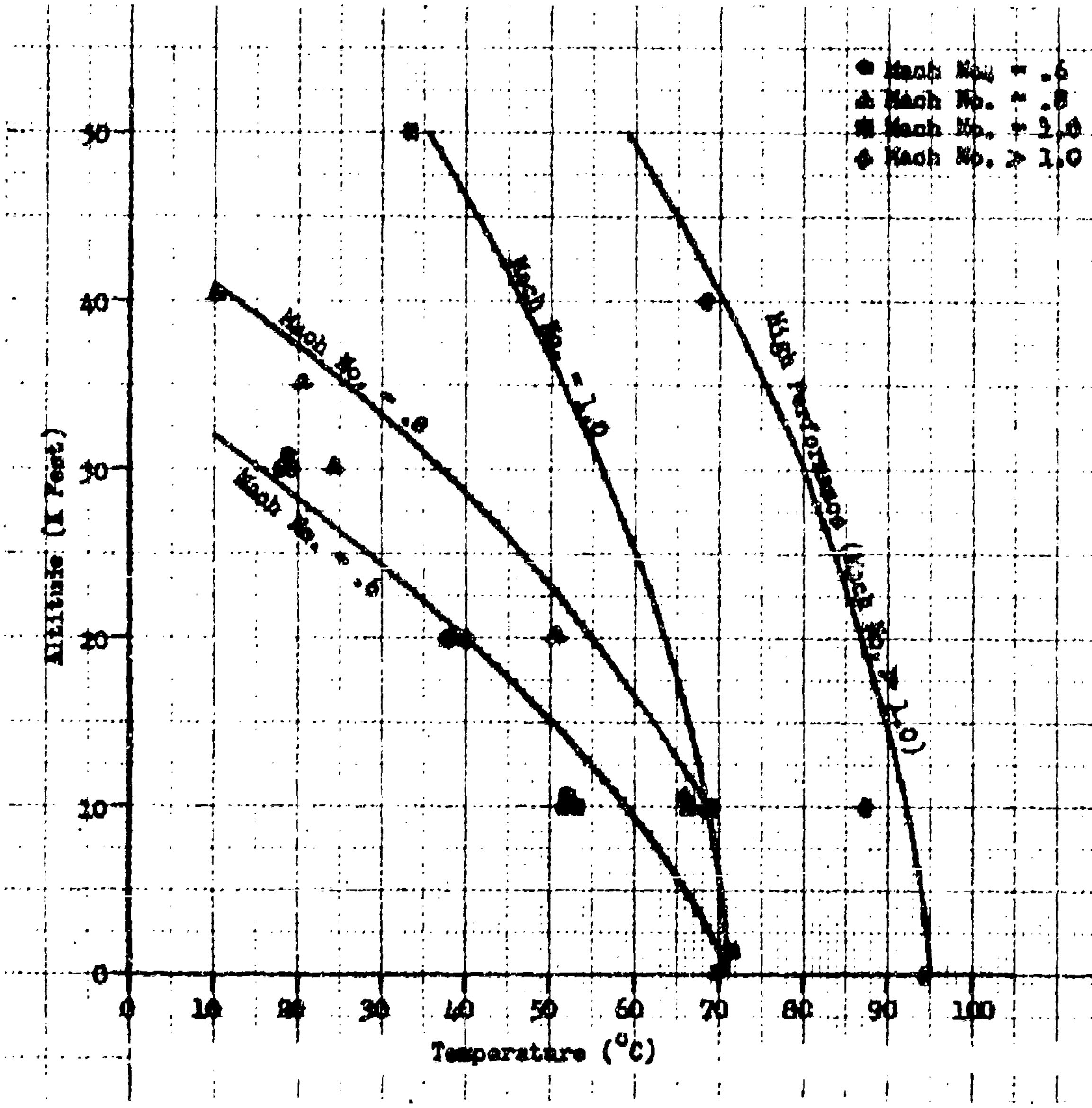


FIGURE 30 CLASS II EQUIPMENT COMPARTMENT TEMPERATURE VERSUS MISSION PROFILE

TABLE 20 HOT DAY COMPARTMENT AMBIENT TEMPERATURES ($^{\circ}C$) FOR CLASS II EQUIPMENT

Altitude (K feet) \ Mach Number	≤ 0.6	0.8	1.0	High Performance*
0	71	71	71	95
10	56	68	68	93
20	40	55	63	88
30	15	36	56	80
40	5	10	46	70
50	5	10	35	60
60	5	10	24	49
70	5	10	11	35

* Ambient cooled equipment must be turned off for 15 minutes after attaining these temperatures to comply to MIL-E-5400 "Intermittent Operation."

TABLE 21 RAM COOLED COMPARTMENT TEMPERATURES (^{o}C) - HOT DAY

Altitude (K feet) \ Mach Number	0.4	0.6	0.8	1.0
0	48	60	75	95
10	27	38	52	71
20	6	16	29	46
30	-15	-6	7	23
40	-36	-30	-16	-1
50	-30	-19	-7	8
60	-31	-23	-11	4
70	-30	-22	-10	5

TABLE 22 RAM COOLED COMPARTMENT TEMPERATURES (^{o}C) - COLD DAY

Altitude (K feet) \ Mach Number	0.4	0.6	0.8	1.0
0	-44	-37	-25	-11
10	-18	-10	2	19
20	-36	-28	-16	-2
30	-58	-50	-40	-27
40	-59	-51	-41	-28
50	-82	-76	-67	-55
60	-82	-75	-66	-54
70	-65	-58	-48	-35

Depending upon the efficiency of the unit's cooling air heat transfer system, it is conceivable that a WRA's internal temperature time history may be completely independent of the external thermal environment. This condition would, in effect, reduce the developed thermal environment to a constant. The results of the analysis of data, relative to forced air cooled WRA's tends to indicate that this condition may have actually occurred during the studied laboratory tests.

In order to assure that this stabilizing influence does not prevent the correct application of the desired thermally induced stresses, the cooling air temperature/flow schedule must be discretely specified. Since the object is to obtain as many equipment operating temperature reversals as possible per unit test time, the cooling air temperature/flow schedule must be specified to produce the extremes of specification tolerance.

By utilizing the cooling air specification curve end points, i.e., maximum flow-minimum temperature for low temperature cycles and minimum-flow - maximum temperature for high temperature cycles, this objective is attained without risking equipment thermal overstress because the airflow/temperature schedule is within the specification tolerance (refer to Figure 31). All changes in cooling air temperature and flow should be accomplished at the test equipment's maximum capability but in no case shall it exceed three minutes.

If any special flow rates are used for ground operation, then these must also be specified to fall at the end points of the envelope as outlined above.

6.2.4 <u>Humidity Cycle</u>

As previously discussed, the basic thermal cycle is constructed as an analog of the aircraft's operational time-history and as such, maintains a high ratio of equipment on time to test time. Furthermore, due to economic and schedule considerations, the overall test program and its various cyclic parts must be so constituted to allow the use of standard test equipment and, insofar as practical, automated cycling.

In considering various alternate methods of developing the humidity cycle and locating it within the thermal cycle, the prime objectives were

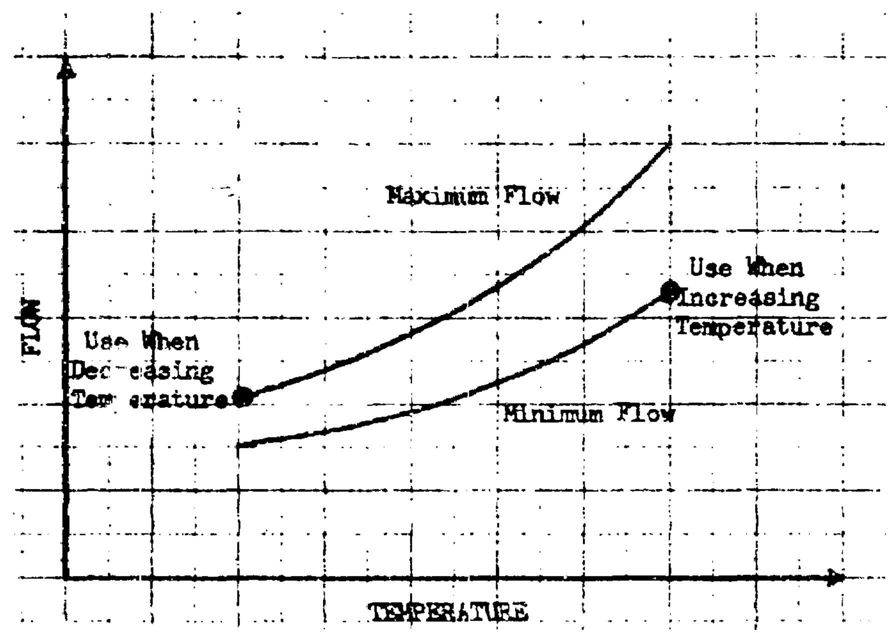
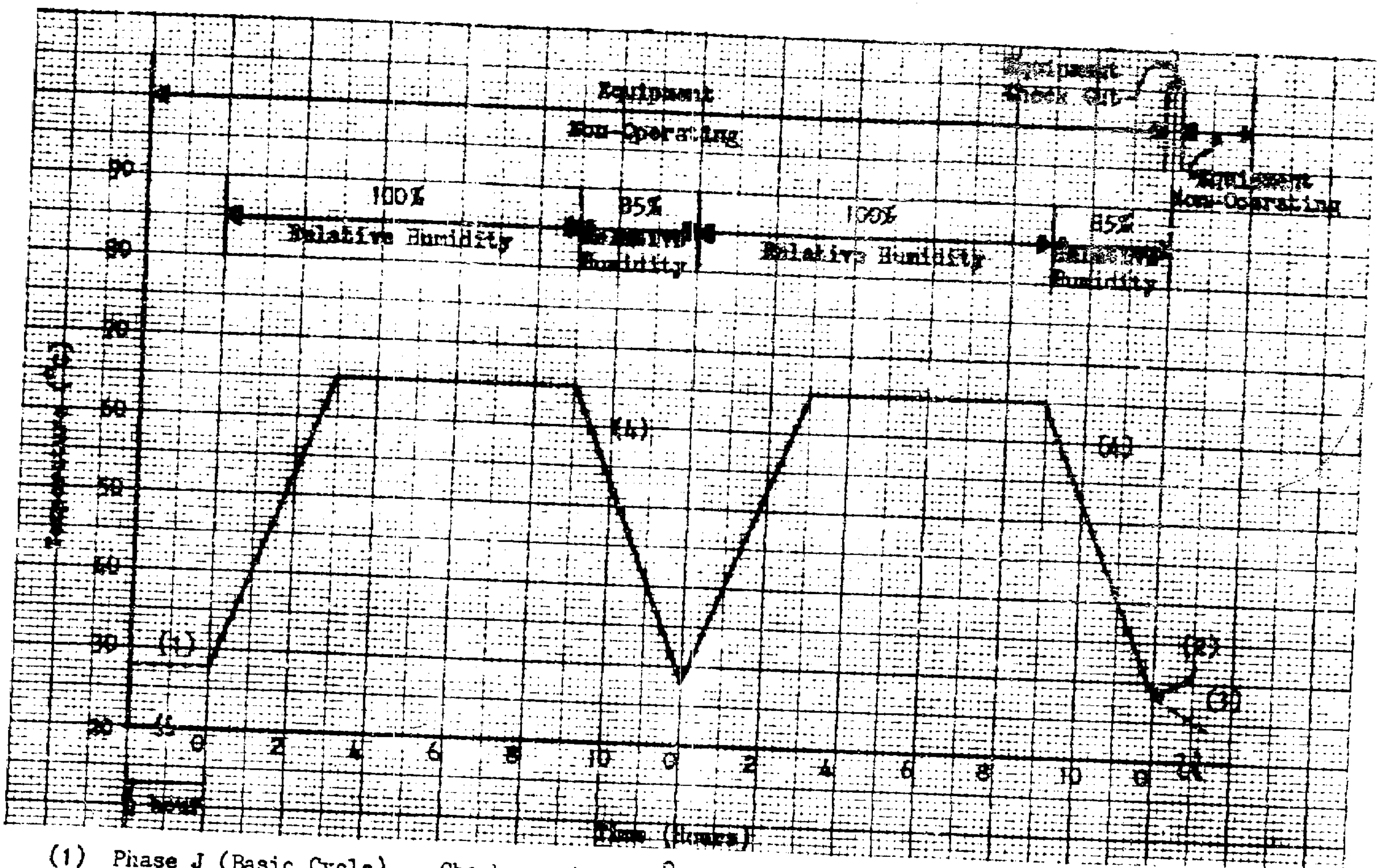


FIGURE 31 TYPICAL WRA COOLING AIR SPECIFICATION

to produce a technically valid exposure whose duration and/or cyclic period fit within the aircraft service life analog concept. Based upon available historical data, it was determined that a technically valid approximation of the average field environment would be achieved if the humidity exposure were represented as 20% of the total test time. It was initially considered that his exposure should occur during the "hot day - non-operating" period (phase E) of the basic test cycle to preserve the mission analog concept. An in-depth study of this placement, relative to the operation of standard test equipment, revealed that substantial changes to the basic profile would be required. In evaluating the technical benefits to be derived from this placement vs. the increase in test complexity and cost, the required justification could not be provided. It was concluded that a far better simulation of the expected natural moisture environment and its effect upon avionics, could be achieved if a variation of the standard MIL-STD-810 exposure were conducted at discrete intervals separating basic cycles.

The proposed humidity exposure (refer to Figure 32) is derived from MIL-STD-810, Method 507, Procedure I. The basic cycle has been modified to increase its efficiency. The number of repetitive cycles during any one exposure has been reduced to allow the exposure to be repeated every twenty thermal test cycles throughout the extent of the test program. This distribution more realistically simulates the field environment than would a long term exposure at any one point in the program.

The method 507, Procedure I standard test cycle has been modified to extend the "rise-to-temperature" period to three (3) hours to assure realization of 100% relative humidity at 65°C for the first cycle. This extension coupled with the standard six hour dwell will afford the greatest opportunity for moisture migration. The drying period, represented by the reduction in temperature to 28°C, has been shortened to a nominal three hour period for test efficiency. The recognized risk (free moisture precipitant within the chamber) associated with reducing the drying time, is minimized by imposing the 85% relative humidity requirement which will, in actuality, govern the duration of this period. Thus, the actual drying time may exceed three hours depending upon the capability of the test equipment to reduce



(1) Phase J (Basic Cycle) — Chamber set at 28°C
(2) Operational check out (Short Duration — GO/NO-GO only) - Equipment off at end of C/O
(3) Reduce chamber temperature to next cycle starting temperature and then dwell for ½ hr. with equipment non-operating
(4) Drying Cycle — Actual duration controlled by 85% relative humidity requirement

FIGURE 32 HUMIDITY PROFILE

the absolute water content.

In order to assure obtaining the full effect of each humidity exposure, while distributing the total exposure throughout the entire test period, each exposure consists of two modified cycles, back-to-back. This arrangement affords two opportunities for the driver constituent (temperature) to have its full effect.

Constructing the humidity exposure and positioning it between basic cycles, as previously outlined, requires that sufficient time be allocated prior to and subsequent to the humidity cycle, to allow the test article to stabilize at the desired initial temperatures. Furthermore, an operational check-out of the test article is considered mandatory at the completion of each humidity exposure. Upon completion of phase I of the basic cycle, the chamber temperature is set at 28°C with the equipment non-operating. At the conclusion of the 1/2 hour, phase "J" dwell period, the average WRA temperature will have stabilized at 28°C, allowing the inception of the humidity exposure. At the conclusion of the humidity exposure, an abbreviated operational check-out of the test article is performed when the chamber temperature reaches 28°C. Following this check-out, the chamber temperature is adjusted to the next thermal cycle "start" temperature and the non-operating equipment is allowed to dwell for 1/2 hours.

6.2.5 Test Level Applicability

The specified test levels which are based upon altitude, velocity and hot and cold day temperature conditions are applicable to the following types of equipment:

(1) Class 1 - Equipment designed for 50,000 feet altitude and continuous sea level operation over the temperature range of -54° to +55°C (+71°C intermittent operation).

(2) Class 1A - Equipment designed for 30,000 feet altitude and continuous sea level operation over the temperature range of -54° to +55°C (+71°C intermittent operation).

(3) Class 2 - Equipment designed for 70,000 feet altitude and continuous sea level operation over the temperature range of -54°C to +71°C (95°C intermittent operation).

(4) Equipment located in a ram cooled compartment.

(5) Equipment cooled using ram air directly or through an air/oil heat exchanger.

6.2.6 Construction of Temperature-Humidity Test Profile

6.2.6.1 Required Information - The development of a laboratory program, directed to a specific aircraft, required the use of certain aircraft/equipment peculiar information in addition to that presented herein. The necessary specifics are as follows:

(1) Flight envelope of aircraft including climb and descent rates, both maximum and idle.

(2) Mission time line of aircraft

(3) Type of equipment to be tested (Class I, Class IA, Class II, equipment located in ram cooled compartment)

(4) Method of cooling. (Ambient, ram air, or forced air) If forced air cooled, a specification curve of flow rate vs. inlet temperature is required.

(5) Minimum expected steady state compartment temperature for cold day.

6.2.6.2 Equipment Operating Schedule

In order to obtain the desired high ratio of operating time to test time, the schedule, which is based upon mission phases and shown in Table 23, should be followed. It should be noted that insofar as possible the schedule durations are derived from the mission profile. The exceptions taken are necessary to obtain the desired environmental exposure within the constraints of test equipment and/or test time.

6.2.6.3 Forced Cooling Air Temperature/Flow Schedule

As previously discussed, the cooling air temperature and flow must be controlled in order to assure that the test article is in fact subjected to the thermal environment exposure. The presented schedule, Table 23 controls the parameters such that the forced air is an aid to the WRA's attaining the desired thermal transitions. The schedule is referenced to

TABLE 23 WRA OPERATING AND COOLING AIR SCHEDULES

PHASE	TEST PHASE DEFINITION	OPERATING/NON OPERATING	AIR FLOW/TEMPERATURE
A	Ground Operation - Cold Day	Operating	Max. Flow/Min. Temp.
B	Climb to Altitude	Operating	Min. Flow/Max. Temp.
C	Mission Objective	Operating	Max. Flow/Min. Temp. (2)
D	Descent to Hot Day	Operating	Min. Flow/Max. Temp.
E	Ground Storage - Hot Day	Non-operating	OFF
F	Ground Operation - Hot Day	Operating	Min. Flow/Max. Temp.
G	Climb to Altitude	Operating	Max. Flow/Min. Temp.
H	Mission Objective	Operating	Min. Flow/Max. Temp. (2)
I	Descent to Cold Day	Operating/Non-operating (1)	Max. Flow/Min. Temp.
J	Ground Storage - Cold Day	Non-operating (1)	OFF

(1) The total non-operating time is desired to be 1 hour (1/2 hour Phase J and 1/2 hour from the total allocated for Phase I.) However, if the total Phase I time is less than 1/2 hour, the total non-operating time will be similarly reduced.

(2) The flow/temperature schedule should be adjusted for this phase (Mission Objective) so that the maximum flow/minimum temperature occurs when the chamber temperature is reduced and the minimum flow/maximum temperature occurs when the chamber temperature is being increased.

the established mission phase.

6.2.7 Sample Test Profile

The following test profile is presented as an aid to the reader's understanding of the proposed approach and its implementation. The required air craft/equipment information is provided, and its utilization in conjunction with the data is shown.

In constructing a thermal test profile, Table 24 forms the base. From Table 24 the fixed durations, i.e., 30 minutes, is obtained for Phases A, E, F, and J. Furthermore, under the appropriate class of equipment will be found either an actual temperature or a Table number to be used in the selection of the temperature for these phases.

Continuing in the aircraft operational phases, one can see that these durations are based upon aircraft mission time lines and the appropriate temperatures are selected from the indicated Tables under the specific class of equipment.

The repetitive period of this thermal cycle and the cyclic insertion of the humidity cycle are indicated in Figure 27. The special conditions associated with the insertion of the humidity cycle are shown in Figure 32.

Construction of Sample Thermal Profile

The "Fighter Intercept" thermal profile is constructed using the following information obtained from the aircraft and WRA performance specifications. The underlined material was used in constructing the profile.

An ambient cooled unit designed for Class II is to be used in a fighter aircraft. The aircraft climbs to 30,000 feet in 7 minutes and is vectored to the target in 23 minutes at a mach number of 1.0 at this time, the fighter makes a high performance dive in 2 minutes to intercept the target at 10,000 feet. After the kill, the fighter cruises at high performance at 10,000 feet for 5 minutes. The fighter then climbs to 40,000 feet in 13 minutes and cruises to base at a mach number of 0.6 cruise time in 35 minutes. Idle descent time is 15 minutes. The steady state compartment temperature is considered to be -18°C at sea level.

TABLE 24 THERMAL PROFILE BASE

PHASE	TEST PHASE DEFINITION	DURATION	CHAMBER LIMITS		
			CLASS I (2)	CLASS II (2)	RAM COOLED (2)
A	Ground Operation Cold Day	30 minutes	Table 18	Table 18	Table 18
B	Take-Off Climb to Altitude	(1)	Table 18	Table 18	Table 22
C	Mission Objective	(1)	Table 18	Table 18	Table 22
D	Idle Let Down & Landing	(1)	Table 19	Table 20	Table 21
E	Ground Non-operation Hot Day	30 minutes	71°C	71°C	71°C
F	Ground Operation Hot Day	30 minutes	71°C	71°C	71°C
G	Take-Off & Climb to Altitude	(1)	Table 19	Table 20	Table 21
H	Mission Objective	(1)	Table 19	Table 20	Table 21
I	Descent & Landing	(1)	Table 18	Table 18	Table 22
J	Ground Non-Operation Cold Day	30 minutes	Table 18	Table 18	Table 18

Notes:

(1) Duration based on aircraft profile.

(2) If chamber "rate of temperature change" limits are exceeded, the chamber limits shall be used. Temperature rate of change shall be 5°C/minute (minimum).

Combining the above information with the direction and information contained in Table I yields that presented in Table 25. The resulting profile is represented graphically in Figure 33.

6.2.8 Profiles for Multi-Mission Aircraft

Where several types of aircraft missions are contemplated, several alternate solutions to the development of a thermal test program for multi-mission aircraft have been investigated as discussed in the following paragraphs.

The solution which would best duplicate the expected environment over a substantial portion of the WRA's life would require testing to as many derived profiles as there are contemplated aircraft missions. The number of exposures to these derived profiles would be in the same proportion as the projected distribution of aircraft missions and the ordering could follow a random selection. The major obvious disadvantages to this approach are cost and complexity. The automated equipment controls would require reprogramming for each profile change, necessitating almost continuous attendance of test personnel during the full extent of the demonstration. Furthermore, documentation for the preparation of the required test plans, procedures and reports would be costly.

The most stringent of the examined alternatives is based upon conducting the demonstration using the most severe of the derived profiles throughout the program. Certainly the advantages to this approach in terms of test cost and complexity are obvious. The disadvantage however, lies in requiring the test article to survive repeated exposures to a level of environmental stress which it may rarely experience during its service life. This type of program could accelerate failures and/or produce failure modes far in excess of actual field performance, where the WRA would be subjected to a variety and mix of environmental levels, thus rendering the demonstration a poor index of expected field performance. Although one could argue that the successful completion of such a test buys a "safety margin" for field usage, (which may be very desirable) the question is "at what price?". In order to assure a product's successful completion of the program, some possible overdesign may be required. Thus the reliability benefits for this

TABLE 25 FIGHTER INTERCEPT THERMAL PROFILE DATA

PHASE	SOURCE	TEST PHASE DEFINITION	TEMPERATURE	DURATION
A	(Given)	Ground Operation Cold Day	$-18^{\circ}C$ (Table 18)	30 min (Table 24)
B	Mission Profile	Climb to Altitude-Cold Day 30,000 feet	$-14^{\circ}C$ (Table 18)	7 minutes
C Mission Objective	Mission Profile	Cruise 30,000 feet	$-14^{\circ}C$ (Table 18)	23 minutes
	Mission Profile	High Performance Dive to 10,000 feet	$10^{\circ}C$ (Table 18)	2 minutes
	Mission Profile	High Performance Cruise, 10,000 feet	$10^{\circ}C$ (Table 18)	5 minutes
	Mission Profile	Climb to 40,000 feet	$-14^{\circ}C$ (Table 18)	13 minutes
	Mission Profile	Cruise to Base	$-14^{\circ}C$ (Table 18)	35 minutes
D	Mission Profile	Idle Descent to Hot Day	$71^{\circ}C$ (Table 19)	15 minutes
E	(Given)	Ground Non-Operation Hot Day	$71^{\circ}C$ (Table 24)	30 min (Table 24)
F	(Given)	Ground Operation Hot Day	$71^{\circ}C$ (Table 24)	30 min (Table 24)
G	Mission Profile	Climb to Altitude-Hot Day 30,000 feet	$56^{\circ}C$ (Table 20)	7 minutes
H Mission Objective	Mission Profile	Cruise 30,000 feet	$56^{\circ}C$ (Table 20)	23 minutes
	Mission Profile	High Performance Dive to 10,000 feet	$93^{\circ}C$ (Table 20)	2 minutes
	Mission Profile	High Performance Cruise 10,000 feet	$93^{\circ}C$ (Table 20)	5 minutes
	(Given)	Equipment off to comply with MIL-E-5400 intermittent operation requirement		
	Mission Profile	Climb to 40,000 feet MACH No. = 0.6	$5^{\circ}C$, (Table 20)	13 minutes
	Mission Profile	Cruise to Base, MACH No. = 0.6	$5^{\circ}C$, (Table 20)	35 minutes
I	Mission Profile	Idle Descent to Cold Day	$-18^{\circ}C$, (Table 18)	15 minutes
J	(Given)	Ground Non-Operational	$-18^{\circ}C$, (Table 18)	30 min (Table 24)

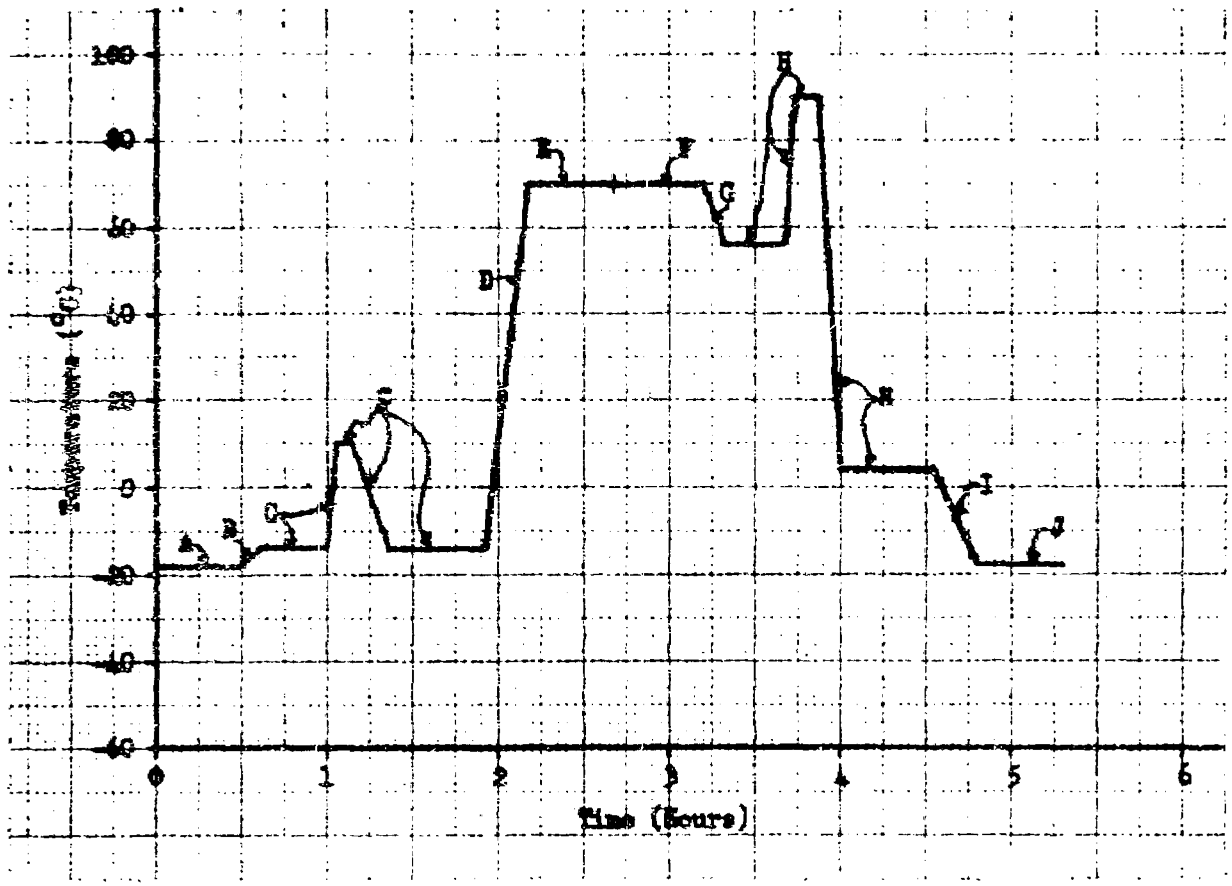


FIGURE 33 PROPOSED THERMAL PROFILE FOR FIGHTER INTERCEPT AIRCRAFT EXAMPLE

approach must be weighed against the additional design, development and recurrent costs.

One solution which retains the cost benefits of a single test profile and provides for the application of typical expected environmental levels was investigated. This approach derives a final test profile as a composite of those derived from the expected missions. The environmental levels and durations associated with each mission segment would be determined as an average of each individual mission derived profile, weighted with respect to the expected frequency of the mission's occurrence. The disadvantage to this approach in that the resultant test profile is in fact an average (albeit a weighted one) which never exposes the test article to the environmental stress levels associated with the more severe missions.

Reviewing the above presented alternatives, one concludes that some compromise approach which retains the major technical and cost benefits of each while minimizing the disadvantages would be an acceptable solution. Given that a mission time line for each of the projected aircraft missions is available and that an accurate prediction of the frequency of occurrence of each of these missions is also available, one can develop a program using environmental severity and frequency of occurrence as biasing parameters. This approach utilizes the most and least severe of the individual mission derived profiles, in a distribution proportional to expected mission frequency of occurrence, to yield a technically valid, cost effective program.

The initial phase of the approach requires that a profile be derived for each projected mission, excluding ferry missions (except for transports), as outlined in paragraph 6.2.6. Each of these derived profiles is then ranked in order of severity which is defined as the number of significant thermal excursions per unit time. A significant thermal excursion being defined as one with a minimum delta of 10°C and a dwell at the new temperature of at least 10 minutes. Assuring the expected "frequency of occurrence" percentage associated with each of these derived profiles completes the necessary data base.

By inspection, one can determine the mid-rank of this list of severity ranked profiles and use it as a dividing line. Summing the "frequency of occurrence" percentages above this line (more severe) and separately summing those below it (less severe) yields two percentages which reflect the expected distribution of levels of environmental severity during the life of the aircraft. The thermal test program can then be constructed utilizing the most and least severe profiles in the derived distribution.

Once having determined the two test profiles and their distribution ratio, the temperature/humidity sequence is conducted as previously described and shown graphically in Figure 27. The distribution of the two profiles within the twenty cycle set should be such that at least one complete ratio-set of profiles is conducted during each week of test time. (i.e., If the distribution ratio is 60% least severe profile and 40% most severe profile, six least severe profiles and four most severe profiles must, as a minimum, be conducted every test week.) In special cases of equipment/systems with very short MTBF's, this minimum requirement may have to be modified to insure a valid distribution of the profiles within the extent of the demonstration.

6.3 VIBRATION

6.3.1 Approach

It is apparent, and can be verified by any study of mission profiles, that almost all of the time during which the equipment of interest is required to meet specification performance is spent under steady state conditions. It is reasonable to assume then that a Reliability Demonstration Test Profile should be based on steady state conditions; the short duration transient situations being adequately covered by successful completion of the environmental Qualification Test.

During the Qualification Test, the equipment is exposed to accelerated test levels at sinusoidal and/or random energies, as well as accelerated half sine shock pulses for short periods of time, so as to demonstrate: (1) the equipment's performance at extreme conditions, which are far in excess of any operational steady state levels and more severe than any transients the equipment will encounter during its operational life, and (2) to evaluate the structural integrity of the equipment construction.

Additionally, since the test level is increased, it is possible by utilizing Stress versus Cycles (S-N) theory, to reduce the test time, and still satisfy the structural design requirements of the equipment. However, at no time during the Qualification testing have the equipment's electrical components been exposed to any long periods of environmental testing.

In regard to the test time associated with the transient conditions, i.e., catapult, buffet, abrupt maneuvers, arrested landing, etc., it is noted that the significant frequencies associated with the transients are generally low, i.e., less than 50 Hz, and are related with the major structural modes. These structural modes i.e., fuselage vertical bending, wing bending, etc., have relatively high displacement inputs to the equipment, but produce insignificant damage because the transient frequencies are generally below the equipment's resonant frequencies, which therefore cause the equipment to displace without producing any dynamic amplification. The number of transient occurrences are at a level approximately 2 times the operational level, and only represent approximately 5% of the total time or

less, depending on the aircraft mission. Therefore, it can be concluded that the combined accelerated vibration and shock levels imposed on the equipment during the environmental qualification does insure the operational system compliance in the aircraft.

Thus, if all the pre-Reliability Demonstration Testing has been completed successfully, i.e., subjecting the equipment to short duration, high level extremes typical of all the transients, only the remaining flight conditions representative of the steady state, long duration environment, should be utilized to determine the Mean Time Between Failures (MTBF) characteristics of the equipment.

Section V indicated the necessity for the demonstration test vibration environment to approximate field conditions both in level and frequency. Therefore the field vibration data presented in Appendix C was analyzed with a goal toward developing representative and comprehensive values of vibration levels and frequency for demonstration testing.

The approach in the analysis of the data was to develop an envelope that generally bounded the data points under review. The upper limit of the envelope was determined through the application of "Statistical Tolerance Limit" techniques (ref. 14). This method provides the means for obtaining an interval which covers a fixed proportion of the population with a specified confidence. The interval is called a "tolerance interval" and the end point is called a "tolerance limit." For this analysis, the confidence was set at 99% and the proportion set at 99.9%. These values were deliberately selected to be conservative. Since the study concentrated on four aircraft, yet the study goal was to have results as widely applicable as possible, the conservatism in approach was considered warrented.

The method of constructing the tolerance limit consisted of determining the mean ($\bar{x}$) and standard deviation (S) of the data points under review and then evaluating the expression:

$$\text{Tolerance Limit} = \bar{x} + kS$$

where:

k is a tabulated value (ref. 14)

ased on

- number of data points
- desired population proportion
- specified confidence.

As previously indicated, the environment experienced by WRA's installed in jet aircraft was different than that experienced by WRA's installed in propeller driven aircraft. Thus, it was decided to develop separate profiles for jet aircraft and turboprop aircraft. The procedures utilized and the results are explained in the following paragraphs.

6.3.2 Levels

6.3.2.1 Jet Aircraft

An initial examination of the jet aircraft vibration data presented in Appendix C, Figures 1 to 14, immediately revealed three primary conclusions, i.e.,

(1) the environment was predominantly random and can not be correctly represented by the MIL-STD-781 fixed sinusoidal frequency requirement.

(2) the internal equipment vibration levels were higher in the rear of the aircraft than in the forward portion.

(3) the data was steady state with no transient responses.

Motivated by conclusion No. 2 above, the next step in the development of the test profile was to examine the vibration environment in each of the aircraft to determine if it was possible to consistently group the WRA's, by levels, into general categories based on location throughout the fuselage. Other factors to be considered were: (1) standardization of vibration level, (2) minimum number of zones, (3) equipment location, and (4) practical relocation of equipment during development phase.

A closer examination of the vibration data recorded on the three jet aircraft indicated a considerable variation in the overall level along the fuselage, the severity increasing toward the engine and further increasing aft of the engine. Further investigation revealed the magnitude of the

vibration in the area forward of the engine compartment, the engine compartment and aft of the engine compartment were similar for the three aircraft even though their engine locations were different, and the engine thrust varied. This observation suggested that three zones (forward of engine, engine compartment, and aft of engine) could be a feasible partition. The engine compartment is defined to start at the plane of the engine fan or compressor front face, and end at the plane of its most aft portion, whether it be at the end of the tail pipe or the after burner.

To verify this assumption, the vibration data for all WRA's located in a zone, irrespective of aircraft, was plotted. Examination of the graphs for each zone revealed the great similarity in values within each zone, thus validating the choice of zones.

A review of the level distribution in each zone indicated that the frequency range should be divided into two bands, i.e., 10-100 Hz and 100-500 Hz. The data points for each frequency band, within each zone, was analyzed to determine specific envelope limits. The composite zone plots and evaluated envelopes are presented in Figures 34, 35, and 36. They indicate:

- Zone I -- (Equipment Forward of Engine Compartment)

  Figure 34 represents the steady state internal vibration environment forward of the engine on jet aircraft. The levels are relatively low, regardless of frequency and indicate that a power spectral density (PSD) level of .01 g^2/Hz between 10-100 Hz and .007 g^2/Hz between 100-500 Hz is representative of the operational environment for equipment forward of the engine. The low levels indicate that the predominant engine induced frequencies have been significantly reduced by the damping in the aircraft structure, and are mainly attributed to the aerodynamic pressure fluctuations impinging on the fuselage.

- Zone II -- (Equipment in Engine Compartment)

  Figure 35 represents the steady state internal vibration environment for the engine compartment in jet aircraft. The PSD level is .002 g^2/Hz between 10-100 Hz and .035 g^2/Hz for the 100-500 Hz

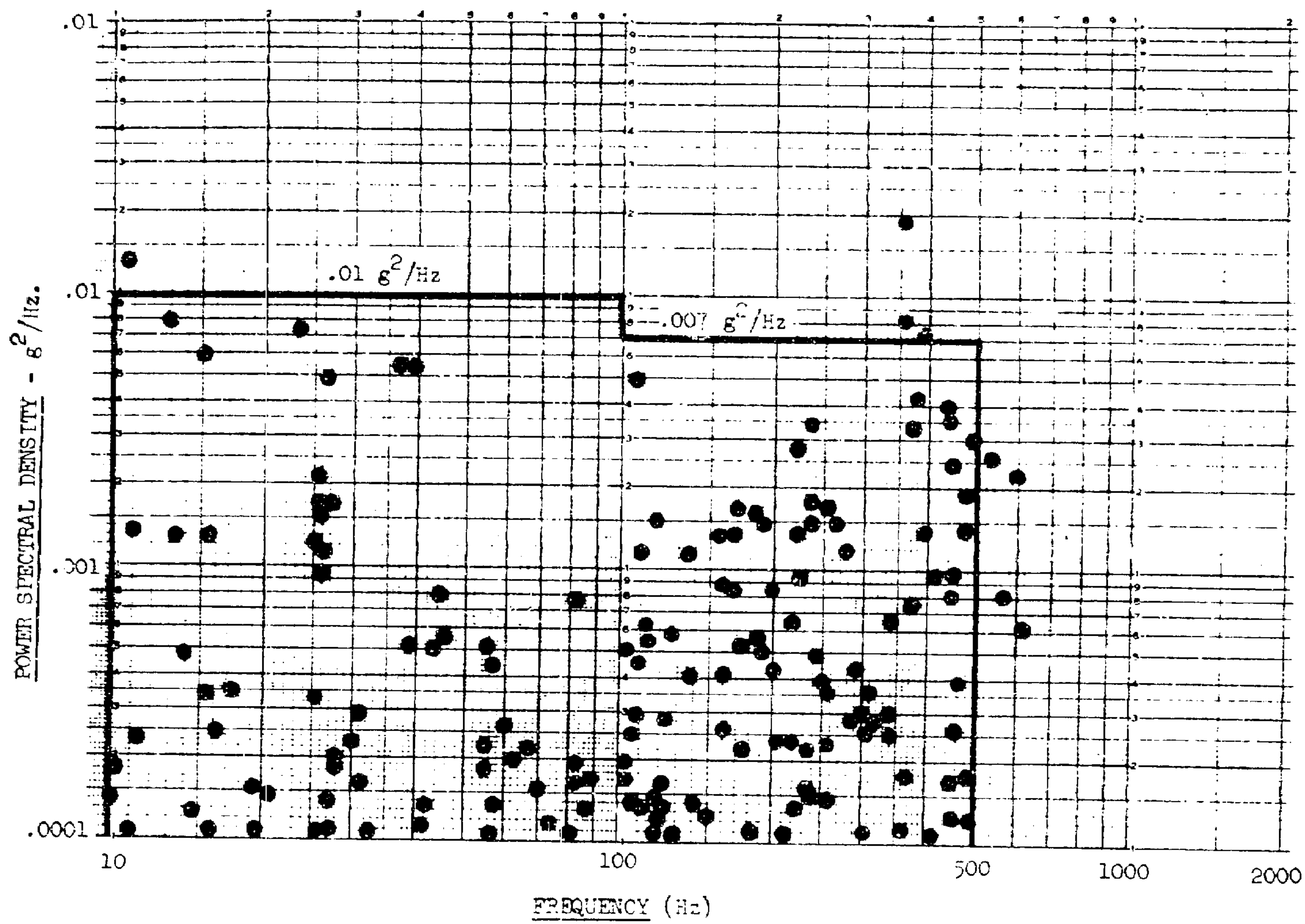


FIGURE 34 COMPOSITE VIBRATION DATA FOR WRA'S INSTALLED FORWARD OF THE ENGINE (ZONE I)

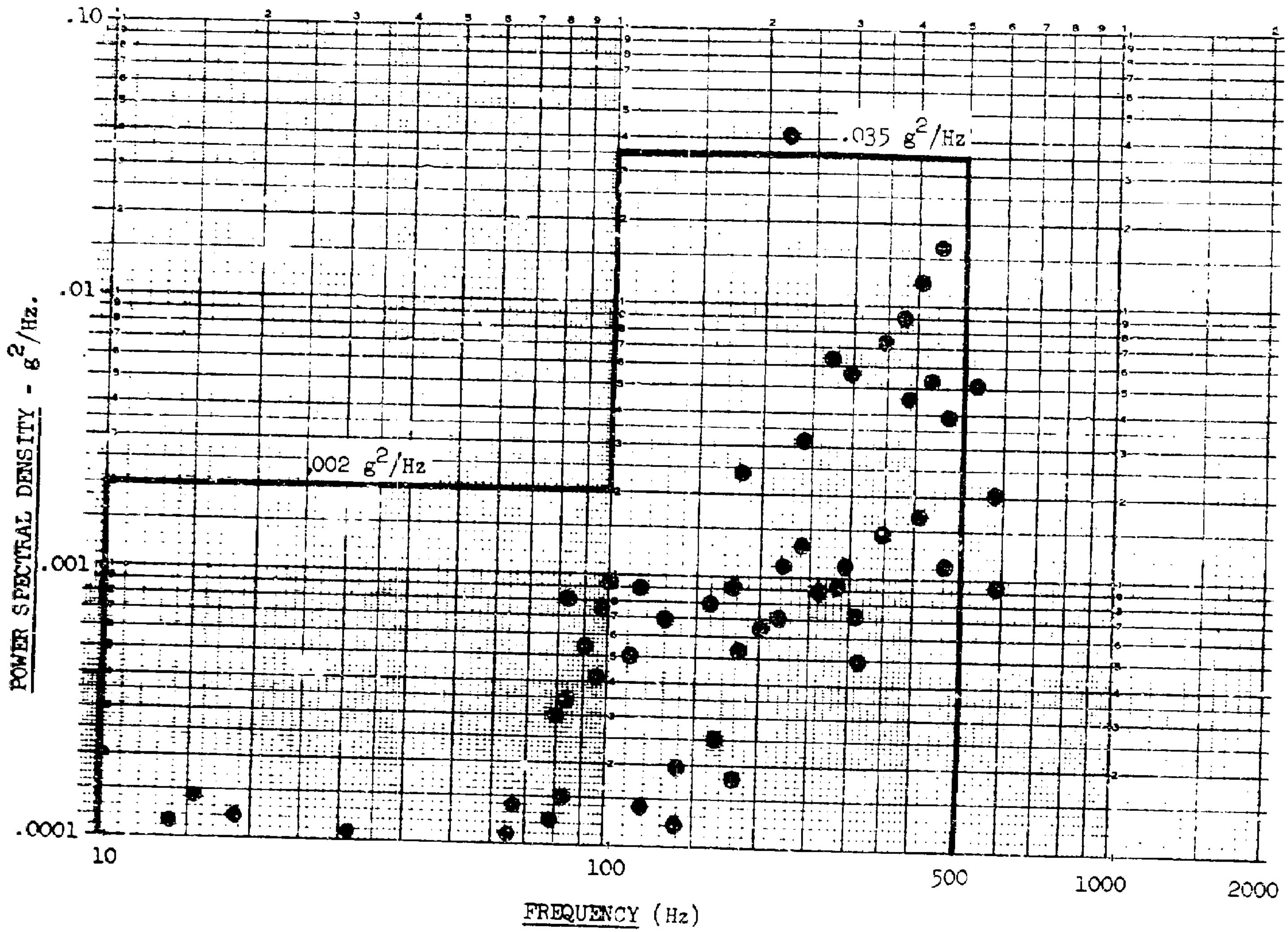


FIGURE 35 COMPOSITE VIBRATION DATA FOR WRA'S INSTALLED IN THE ENGINE COMPARTMENT (ZONE II)

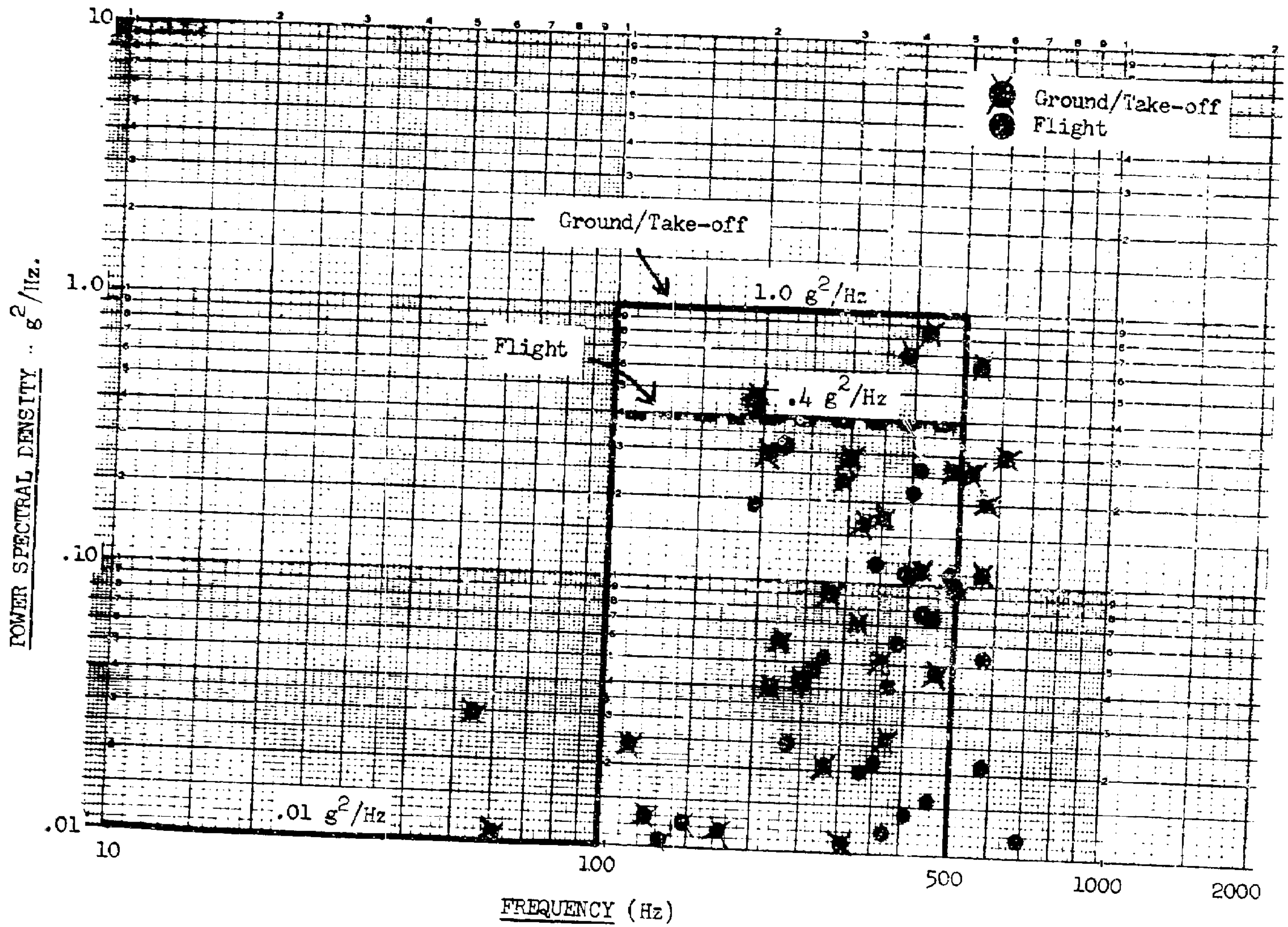


FIGURE 36 COMPOSITE VIBRATION DATA FOR WRA'S INSTALLED AFT OF THE ENGINE COMPARTMENT (ZONE III)

range. The data levels indicate that there is very little displacement at the lower frequency structural modes, but in the 100-500 Hz range the engine rotational frequencies and their harmonics have become significant.

- Zone III -- (Equipment Behind Engine Compartment)

Figure 36 represents the steady state internal vibration environment for the fuselage area behind the engine for jet aircraft. Since the measurements acquired during ground/take-off were generally considerably higher than flight measurements, it was decided to determine separate envelopes for each condition.

The ground levels between 100-500 Hz are extremely high ($1.0g^2/Hz$) and are attributed to engine induced vibrations and reflection off the ground, i.e., direct engine exhaust impingement on the aircraft structure and exposure due to the acoustic field generated by the engine jet exhaust. The flight levels in this frequency range are $.40g^2/Hz$ and are as previously mentioned, primarily the result of engine exhaust impingement on structure and the engine generated acoustic field, but of course are lesser than the ground levels due to the reduction caused by the forward speed of the aircraft, and the elimination of any ground reflections.

It should be noted that in choosing the envelope of $.01g^2/Hz$ in the frequency range of 10-100 Hz an exception to the statistical approach was chosen. Since this frequency range of 10-100 Hz had so few data points it was decided to utilize data from Zone I which had a substantial data population. The rationale for this decision was that low frequency responses are primarily due to structural modes (e.g., fuselage bending, fuselage torsion, etc.) affecting the entire fuselage. A high impedance structure (i.e., engines) is in Zone (II) that does not respond to these low frequency inputs but acts more like a fulcrum. Thus the zones forward and aft of the engine (Zone I and Zone III) respond similarly with respect to each other. It is for this reason that data for Zone I from 10-100 Hz is considered equivalent to the Zone III data.

The two points falling above this $.01g^2/Hz$ were attributed to erroneous instrumentation.

Since the vibration data analyzed represented the operational environment only for the frequency range from 10 to 500 Hz for the three study jet aircraft, an investigation was initiated to determine how representative the zonal concept and calculated envelopes were of other aircraft. Data representing the Zone I and II vibration environment in several additional jet aircraft (ref. 15 and 16) was analyzed to determine if the measured frequencies and associated levels fell within the calculated zonal envelopes. Each measurement location was examined and the vibration data was plotted in its designated zone. This new data contained frequency information out to 2000 Hz and since the frequency ranges in all the new equipment procurement specifications have been extended to 2000 Hz, it was decided to incorporate this information in the investigation.

Examination of this new vibration data showed that from 70-500 Hz, over 95% of the data fell within the envelope limits, indicating excellent agreement and thus further substantiating the use of the vibration zones. Data below 70 Hz was not obtainable due to the excessive bandwidths of the analyzer filter. It was decided that because of the excellent correlation in the vibration data from Reference 15 and Reference 16 up to 500 Hz, it was technically feasable to use the data above 500 Hz to determine the high frequency envelop limits for these two zones. No Zone III data other than for the study aircraft was obtainable. Therefore, available measurement information in the high frequency region on these aircraft was used to extend the profiles to 2000 Hz for this zone. This additional vibration data and its relationship to the previously calculated envelopes are shown in Figures 37, 38, and 39 for Zones I, II, and III, respectively. It should be noted that all the data presented in Figure 39 represents ground and take-off conditions. Therefore, in order to arrive at an inflight high speed level for the 500 to 2000 Hz range, the ratio of ground to flight level observed in the 100 to 500 Hz range, (i.e., 2.5:1) was utilized to produce the level of $0.68g^2/Hz$. These envelopes are then the recommended vibration levels for demonstration testing. It should be noted that PSD transitions are indicated as step functions in order to encompass the full

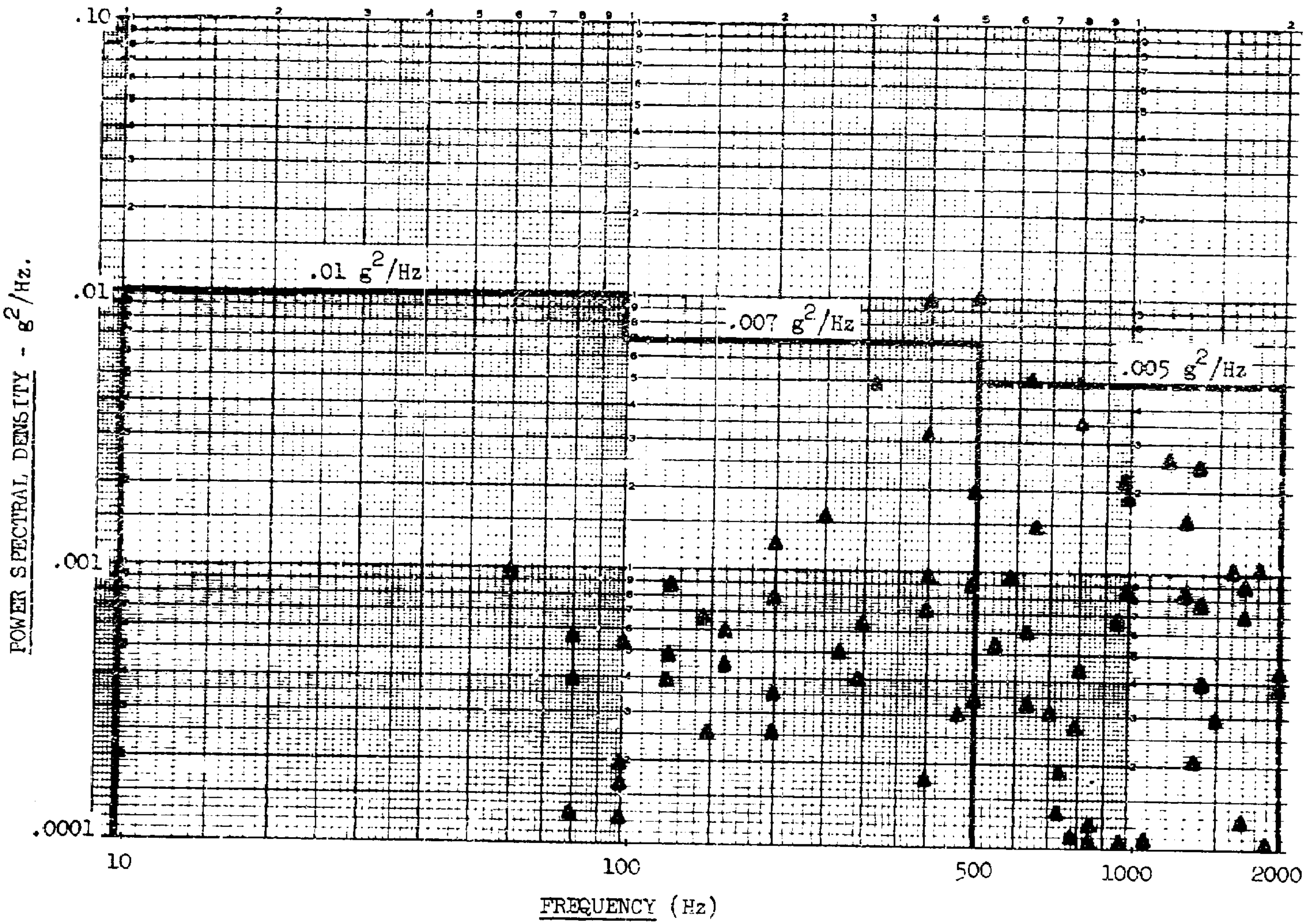


FIGURE 37 VIBRATION DATA ON ADDITIONAL AIRCRAFT STUDIED FOR WRA'S LOCATED FORWARD OF THE ENGINE (ZONE I)

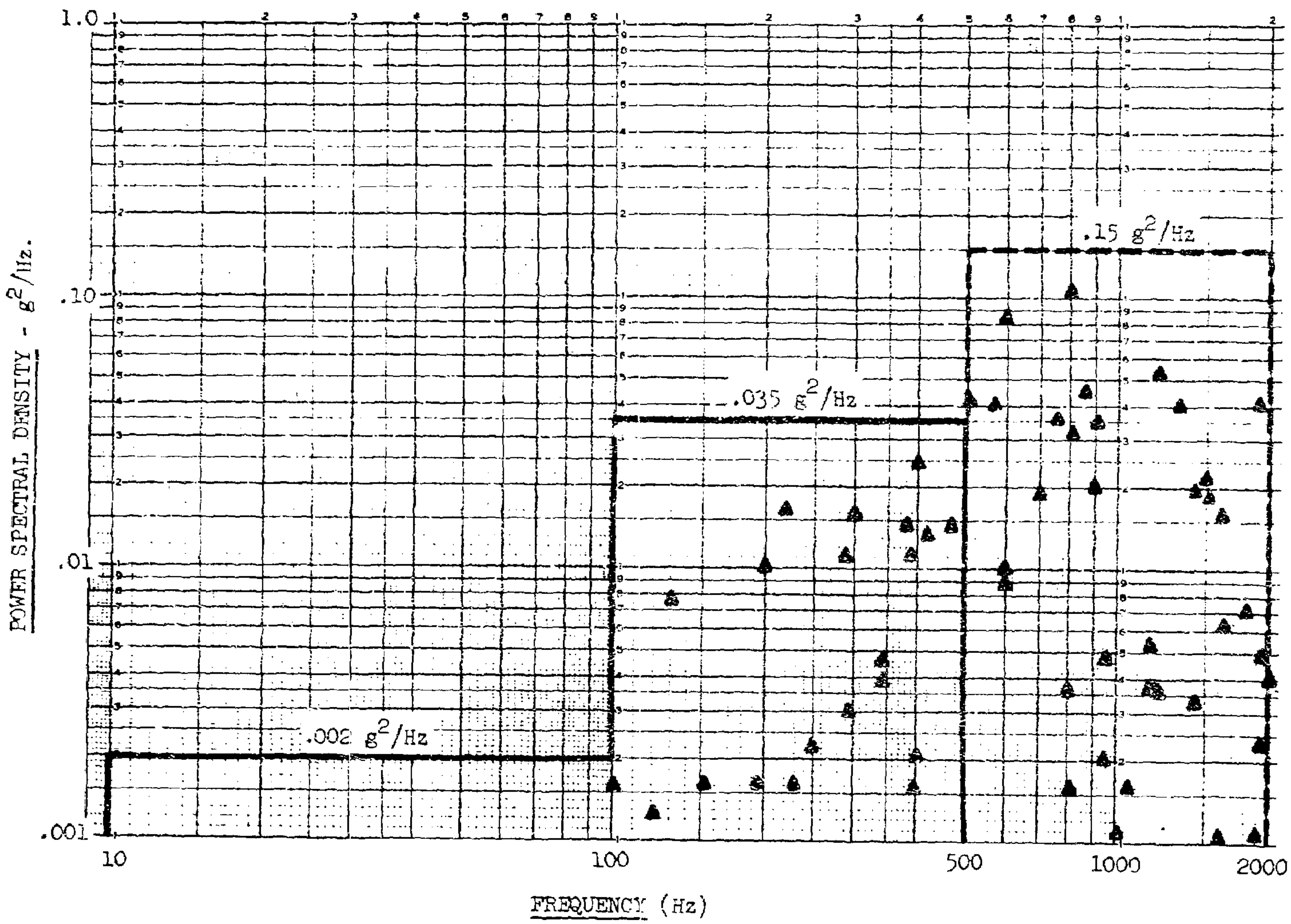


FIGURE 38 VIBRATION DATA ON ADDITIONAL AIRCRAFT STUDIED FOR WRA'S LOCATED IN THE ENGINE COMPARTMENT (ZONE II)

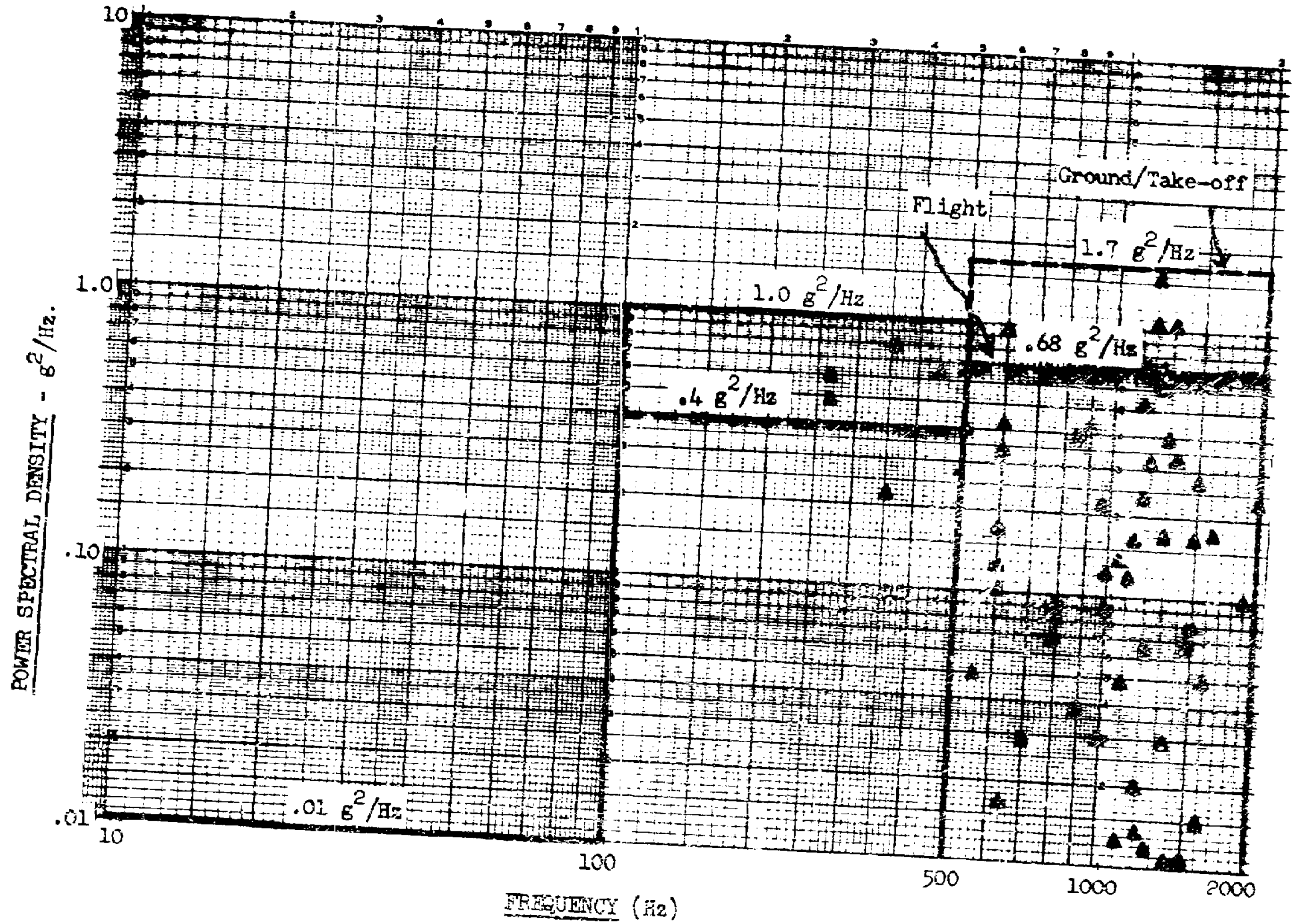


FIGURE 39 ADDITIONAL VIBRATION DATA ON STUDY AIRCRAFT FOR WRA'S LOCATED AFT OF THE ENGINE (ZONE III)

range of data points. Actual testing will require the use of the highest dB/octave ratio available from the test equipment.

6.3.2.2 Turboprop Aircraft

As indicated previously, only one turboprop aircraft was included in the study. This aircraft has two constant speed turboprop engines, one mounted on each wing. An examination of the measured vibration environment, presented on acceleration (g) peak versus frequency plots, as shown in Figures 15 to 17 of Appendix C, describes a vibration distribution indicative of a negligible broadband random base with a series of high (g) peak narrow band spikes. The predominant frequency content is sinusoidal and is directly associated with the engine propeller shaft frequency (18.4 Hz), propeller blade passage frequency (73.2 Hz), and their harmonics. The maximum acceleration (g) responses are concentrated at 73.2 Hz, 146.4 Hz, 219.6 Hz, and 292.8 Hz. The higher frequency energy in the fuselage that is associated with the engine turbines is reduced by the engine low frequency isolator mounts and the structural attenuation in the wing and fuselage.

A review of the above vibration data indicates that the constant-speed turboprop aircraft examined in this study is not completely characteristic of the aircraft group classified "turboprop." As stated previously, all the vibration energy is concentrated at four or five discrete frequencies associated with the propeller and do not vary with flight conditions. An increase in forward velocity is a function of propeller pitch, whereas for the majority of turboprop aircraft, forward velocity is a function of variable RPM and/or blade pitch, i.e., take-off and maximum speed occurs at a higher engine RPM than cruise or loiter. But in either situation, for constant speed turboprop or variable speed turboprop engine aircraft, it can be seen that the operational vibration environment can not be described by the low-fixed frequency (20-60 Hz) requirement outlined in MIL-STD-781.

Since the intent of this study was to develop a widely applicable and realistic requirement, it was decided to analyze several other turboprop aircraft in an attempt to arrive at a universal test spectrum, rather than to have a profile for each and every turboprop aircraft. Thus, the data from the study aircraft as well as the vibration environment on several

other constant and variable speed turboprop aircraft were grouped to examine their frequency and level distribution. Results of this investigation disclosed relatively low acceleration (g) peak responses at frequencies associated with the aircraft structural modes (i.e., fuselage vertical bending, wing bending as well as the frequencies associated with engine shaft rotation), and then higher acceleration (g) responses associated with the localized structure, propeller blade passage and its harmonics as well as frequencies related to the operation of the aircraft auxiliary system pumps and motors. Then, based on experience, and allowing for the variation in minimum operational engine shaft speeds, it was decided to divide the frequency range into two bands, i.e., 10-50 Hz and 50-500 Hz. The data within each band was used to calculate envelope limits. The composite data and resulting envelope are shown in Figure 40. It represents the steady state internal operational environment in the fuselage of turboprop aircraft. The acceleration (g) peak level is ± .7g from 10-50 Hz and ± 2.4g from 50-500 Hz. The predominant responses are at the propeller blade passage frequencies and their harmonics, and vary as a function of engine rotational frequency and number of propeller blades.

The vibration data analyzed in this study represents the steady state environment for the frequency between 10-500 Hz for the study aircraft, in addition to several other constant and variable engine speed turboprop aircraft. The data examined, although limited, indicated that there is significant data above 500 Hz. Based on the high frequency environment evaluated on the jet aircraft and since turboprop and turbofan power plants have certain dynamic similarities by nature of their design, it is recommended that the 10-500 Hz profile be extended to 2000 Hz. resulting in a profile described by ± .7g from 10-33 Hz, .012" DA from 33-62 Hz and ± 2.4g from 62-2000 Hz.

6.3.2.3 Excluded WRA Locations

The preceeding development for internally mounted equipment does not hold true for external and surface mounted equipment installations, that are primarily susceptable to the jet noise and turbulent airflow which impinges on aircraft external surfaces. For these situations, a generalization is

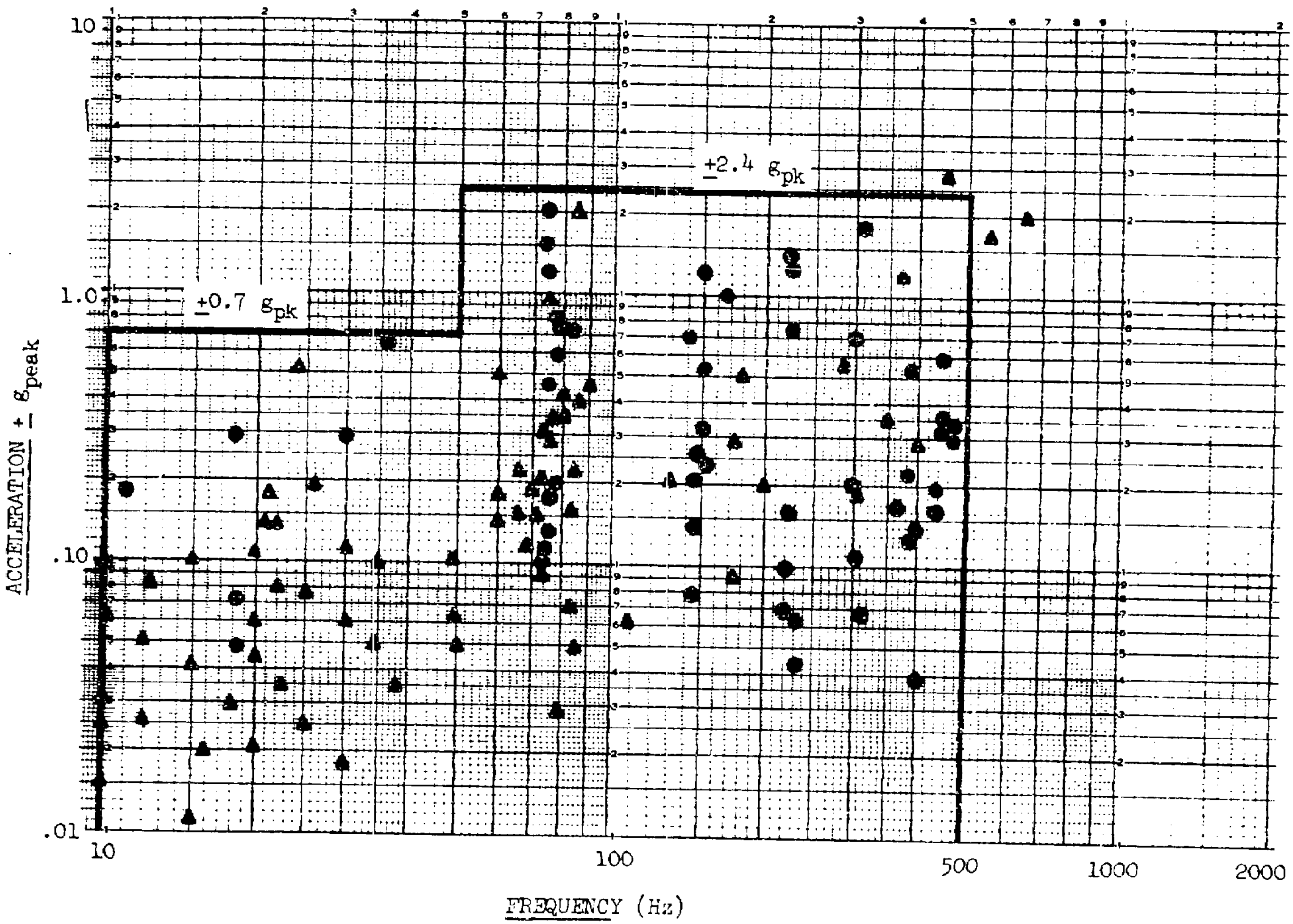


FIGURE 40 COMPOSITE VIBRATION DATA FOR ALL PROPELLER DRIVEN AIRCRAFT INVESTIGATED

not possible and requires a specific knowledge of the engine characteristics, aircraft flight profile and structure to which equipment is mounted. With the above information one can predict the operational surface environment by incorporating the procedures mentioned in Reference 17 for calculating the structural vibrations induced by turbulent airflow. To determine the structural vibrations due to engine noise, one must first calculate the sound pressure levels using Reference 18, and then convert them into frequencies and levels by incorporating the procedures outlined in Reference 17. The resulting levels from above method should then be converted to PSD levels (based on analysis filter bandwidth) and enveloped in such a way so as to encompass all the maximum responses. The resulting envelope will then define the demonstration test vibration level.

6.3.2.4 Extension to Other Aircraft

A review of several different types of aircraft concluded that the vibration environment in jet aircraft is quite similar and is primarily due to the location of the engine relative to the equipment and the flight profile of the aircraft. This is supported by an empirically derived conclusion that:

"aircraft structure selection and construction is proportional to engine thrust and flight envelope. In other words, the design requirements imposed by mission, gross weight, speed, etc., appearing at comparable locations differ little from aircraft to aircraft regardless of type and size. This suggests a relationship of the type

(THRUST) x (STRUCTURAL ATTENTUATION) $\approx$ (Constant)"

Since the commonality of the dynamic environment in aircraft exists, the profiles developed herein will adequately reflect the environment for any internally mounted equipment installed in an aircraft whose engines are fuselage mounted.

It was also determined from a review of available data that the vibration environment for internal fuselage mounted equipment is more severe in aircraft with engines in the fuselage, than in aircraft with engines mounted on the wings or in external pods. Recorded measurements for the latter cases were generally observed to be no greater than $0.01g^2/Hz$ throughout the

fuselage. Intuitively, this lesser level is easy to comprehend, since the engine, the primary disturbing energy source in the aircraft is separated from the aircraft, resulting in a reduction in the mechanical vibration level due to structural attenuation and a reduction in the acoustical level as a result of distance.

Therefore, realizing the benefit of a reduction in vibration level, as well as examining the Reliability Demonstration Profiles presented herein, it is recommended that the reliability test vibration profile that best describes the operational environment for equipment installed in aircraft with engines mounted on the wings or in external pods, is that of Zone I (Fig. 37).

Furthermore, it is estimated that the equipment evaluated in this study is representative of approximately 90% of all current equipment installations in aircraft. Thus, the reliability test vibration profiles developed in this study encompass 90% of all aircraft equipment installations.

As indicated previously, the vibrational characteristics of the remaining installations (i.e., internal and/or external surface mounted) cannot be represented by an environment that was developed for internally mounted equipment. The various techniques utilized to predict the reliability test level for these special cases are discussed in the following paragraph.

6.3.2.5 Prediction of Test Levels

For those situations in turbojet aircraft where the dynamics engineer desires to calculate the operational equipment vibration environment using various present day prediction techniques, the following are available:

(1) Noise Prediction Techniques:

a) AFFDL-TR-62-26 -- jet engine noise at a desired location (Reference 19).

b) AFFDL-TR-71-63 -- jet exhaust noise for ground run-up and flight (Reference 20).

c) AFFDL-TR-67-167 -- boundary layer pressure fluctions (Reference 21).

(2) Vibration Prediction Techniques:

a) WADC-TR-58-343 -- flight vehicle noise predictions (Reference 22).

b) AFFDL-TR-71-63 -- APPENDIX V - response to aeroacoustic excitation (Reference 20).

c) S & V Bulletin No. 28 August 1960 -- vibration levels in jet powered vehicle (Reference 18).

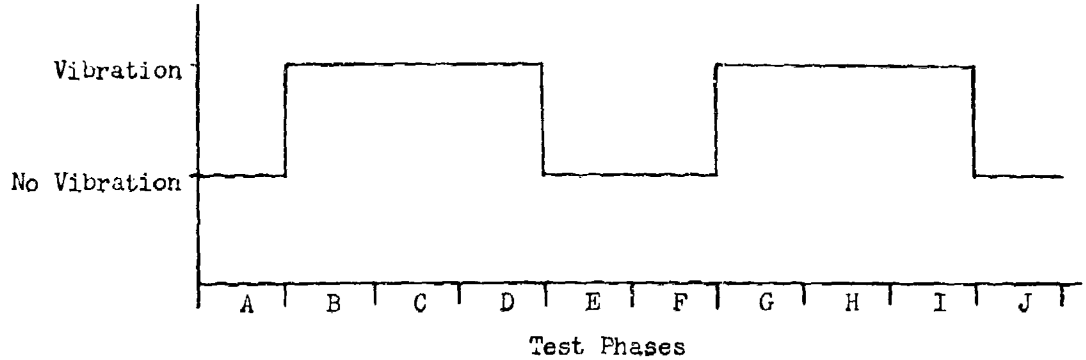
After acquiring data from above methods, the levels should then be converted to PSD levels and the points should be enveloped such that the maximum points in the spectrum are covered. The resulting envelope will provide the demonstration test profile.

No known analogous approach exists for turboprop aircraft. The test levels developed herein appear comprehensive in that when compared with measurements on several other turboprop aircraft, the levels for the same flight conditions fell within the proposed profile. This seems reasonable, since the engines are mounted on the wings, where the benefit of structural attenuation is present and the relationship between the propeller and the fuselage is relatively the same.

6.3.3 Test Durations

Section V indicated that vibration test durations should be increased from the current MIL-STD-781 requirement of 10 minutes of every hour. It was argued that since the WRA is exposed to vibration throughout each flight, the vibration test duration should be proportionately as long. This can best be accomplished by requiring vibration throughout Phases B, C, D, G, H, and I of the basic test cycle described in Paragraph 6.2. Since these phases are the test analogs of a flight and their times are determined from the mission profile of the intended aircraft, the WRA will consequently be exposed to as much vibration as there are simulated missions throughout

the entire test. A simplified graphic representation of this requirement is shown below.



Since the equipment is to be vibrated constantly throughout the flight analog test phases, the accummulated vibration test time could be greater than 1000 hours for higher reliability equipment. Thus, a more realistic examination of the test level in each zone was necessary, to insure that the equipment would not be disproportionately overstressed. As previously mentioned in the study, the vibration data examined for the development of each test zone were acquired during the most severe steady state conditions, i.e., high speed flight (maximum q) and maximum engine power settings during ground operations/take-off and flight. These however, do not represent the operational vibration environment the equipment will experience during the entire aircraft mission. A review of typical aircraft mission profiles indicates that approximately 25 minutes out of any given flight are at these severe conditions with the remaining steady state flight time spent at the more benign levels associated with the cruise and loiter portion of the mission. Furthermore, it was determined from this review that, on the average, two of these 25 minutes are spent during ground operations and take-off.

Examination of measured in-flight vibration data for the cruise and loiter conditions, indicates that they can be adequately represented by a test level that is 50% of the minimum flight levels previously determined for the frequency range in each zone, i.e., (25% of the PSD level for random and 50% of the acceleration or displacement values for sine). Exceptions to this 50% reduction rule are those WRA's requiring the use

of the Zone I profile. These include:

- WRA's installed forward of the engine compartment for those jet aircraft where the engine(s) are mounted in the fuselage
- WRA's installed in jet aircraft where the engines are on the wing or in external pods,

where the data indicates that there is a negligible change in vibration levels throughout all the steady state conditions.

Thus, to reasonably approximate field conditions (for those WRA's where the reduction in level applies) it is recommended that the vibration test be conducted at maximum levels for 25 minutes of each flight analog (i.e., during test phases B, C, and D and then again during test phases G, H, I) with the remaining vibration time at 50% of the maximum in-flight level. To further simulate mission conditions, it is recommended that the test time at maximum levels be apportioned to the phases in the following manner:

- Zone II

  10 minutes at the maximum level to coincide with the start of Phase B (take off and climb to altitude analog)

- Zone III

  Two minutes at the maximum ground level and 8 minutes at the maximum flight, to coincide with the start of phase B (take off and climb to altitude analog)

- Zone II and III

  15 minutes at the maximum flight level to coincide with the "dynamic" (i.e., combat, high speed dash, etc.) portion of Phase C, or 15 minutes midway through Phase C for those aircraft types that do not normally experience "dynamic manouvering" (e.g., transport early warning, etc.)

The above is to be repeated for Phases G and H. As indicated previously, no reduction in level is required for Zone I.

It should be noted that the partition of the 25 minutes at maximum

levels into 10 and 15 minute intervals is applicable to WRA's installed in jet aircraft where the proposed test method is random vibration. For WRA's installed in propeller driven aircraft, where the proposed test method is sine sweep, the recommended partition is 12.5 minutes at each of the phases in the test described above. This modification was necessary to achieve a complete sweep from 10-2000-10 Hz at a sufficiently moderate sweep rate (1.22 octaves/minute) to assure adequate exposure at all frequency bands.

Thus, the vibration test is commensurate with mission time and anticipated level for the aircraft in which the WRA is to be installed.

6.3.4 Application

Figures 41, 42, 43, and 44 present the recommended vibration test envelopes for the equipment location/propulsion type combinations previously described. It is further recommended that the WRA's intended aircraft orientation and mounting (i.e., hard or isolated) be duplicated in the vibration test. As indicated previously, the vibration test profiles for reliability demonstration testing developed herein are applicable for all internally mounted avionics equipment. The profiles have been developed with the objective of approximating, as closely as possible, the vibration environment a WRA is expected to experience in actual field use. Thus, proper application of these profiles requires foreknowledge of those factors, which the analysis indicates principally determines the field vibration environment. Specifically, these include:

- type of propulsion of the intended aircraft
- location of the engines
- location of the WRA relative to the engines
- mission profile timeline

Table 26 shows the specific vibration test requirements for the applicable variations of this information.

6.4 ALTITUDE

It is recommended that altitude not be included as a demonstration test environment. The arguments to support this position are:

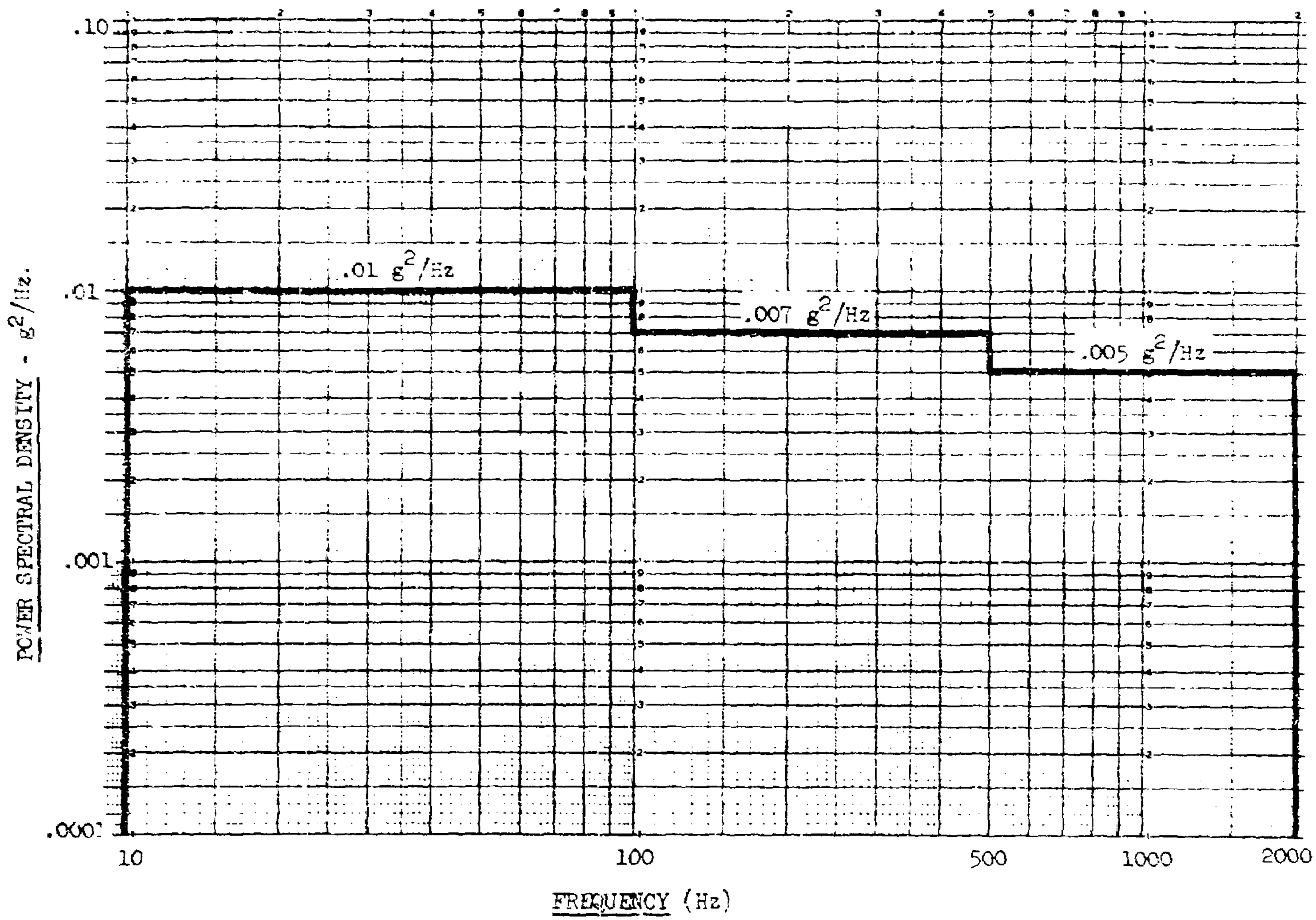


FIGURE 41 RECOMMENDED VIBRATION TEST ENVELOPE -- ZONE I (FORWARD OF ENGINE)

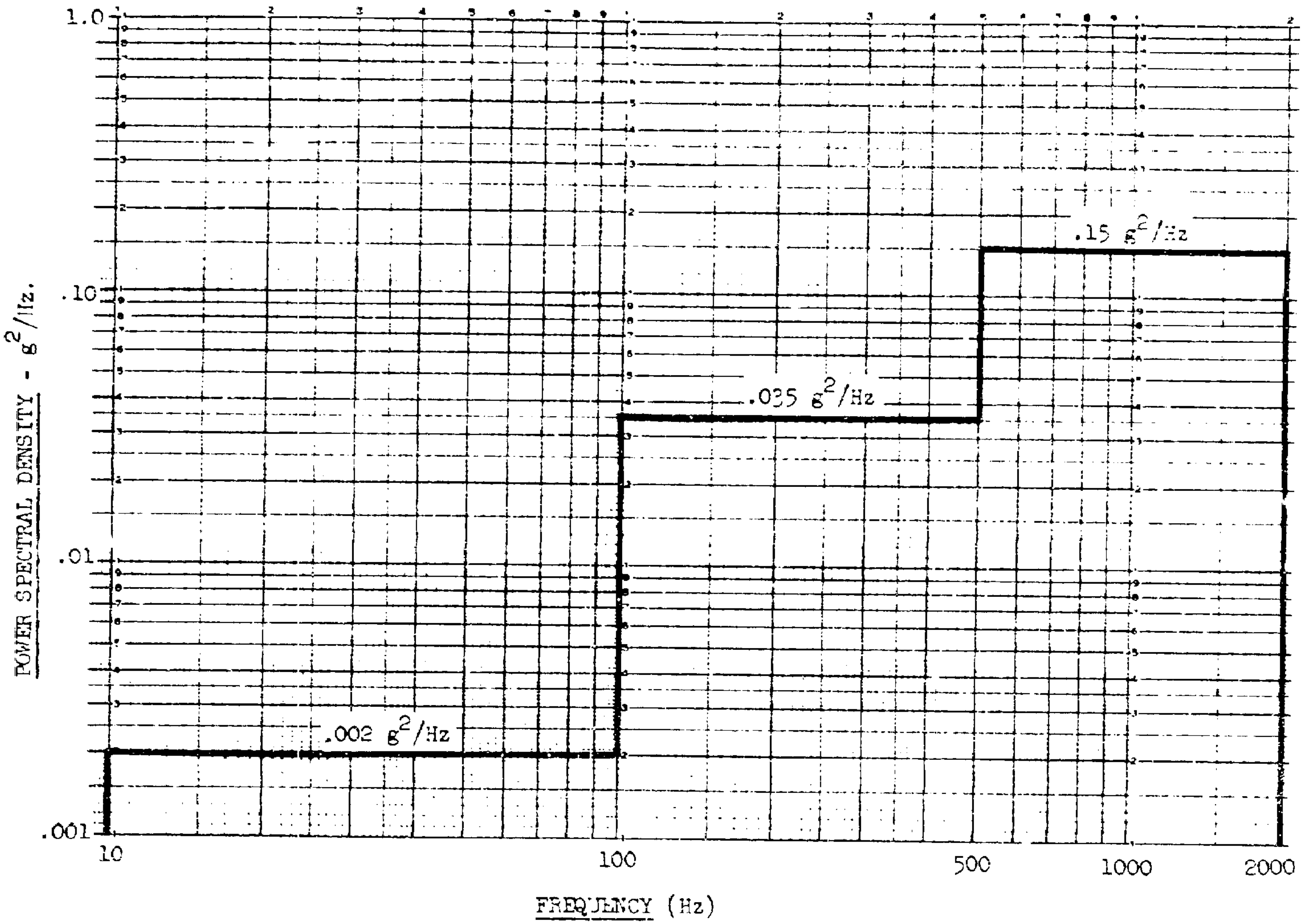


FIGURE 42 RECOMMENDED VIBRATION TEST ENVELOPE -- ZONE II (ENGINE COMPARTMENT)

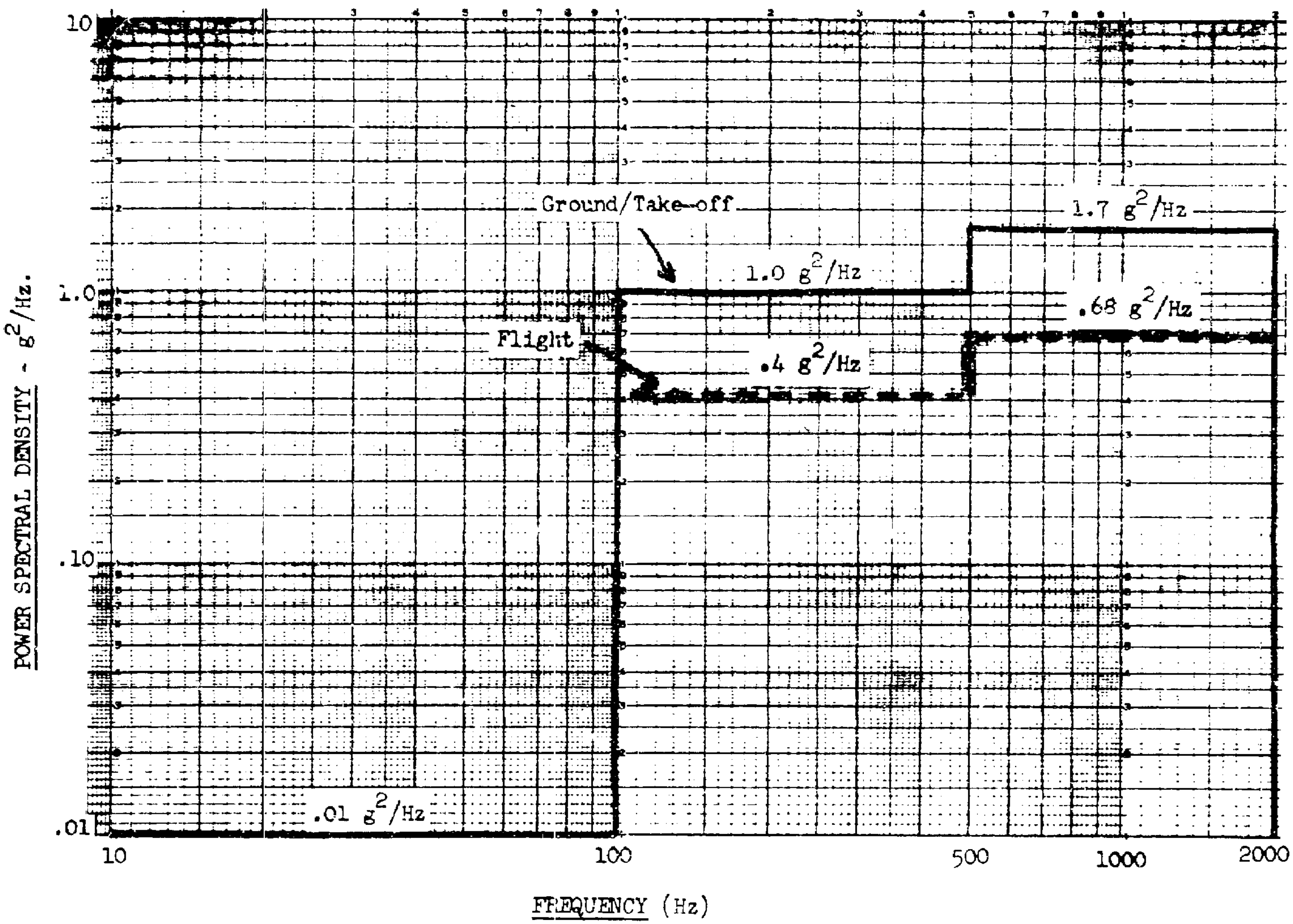


FIGURE 43 RECOMMENDED VIBRATION TEST ENVELOPE — ZONE III (AFT OF THE ENGINE)

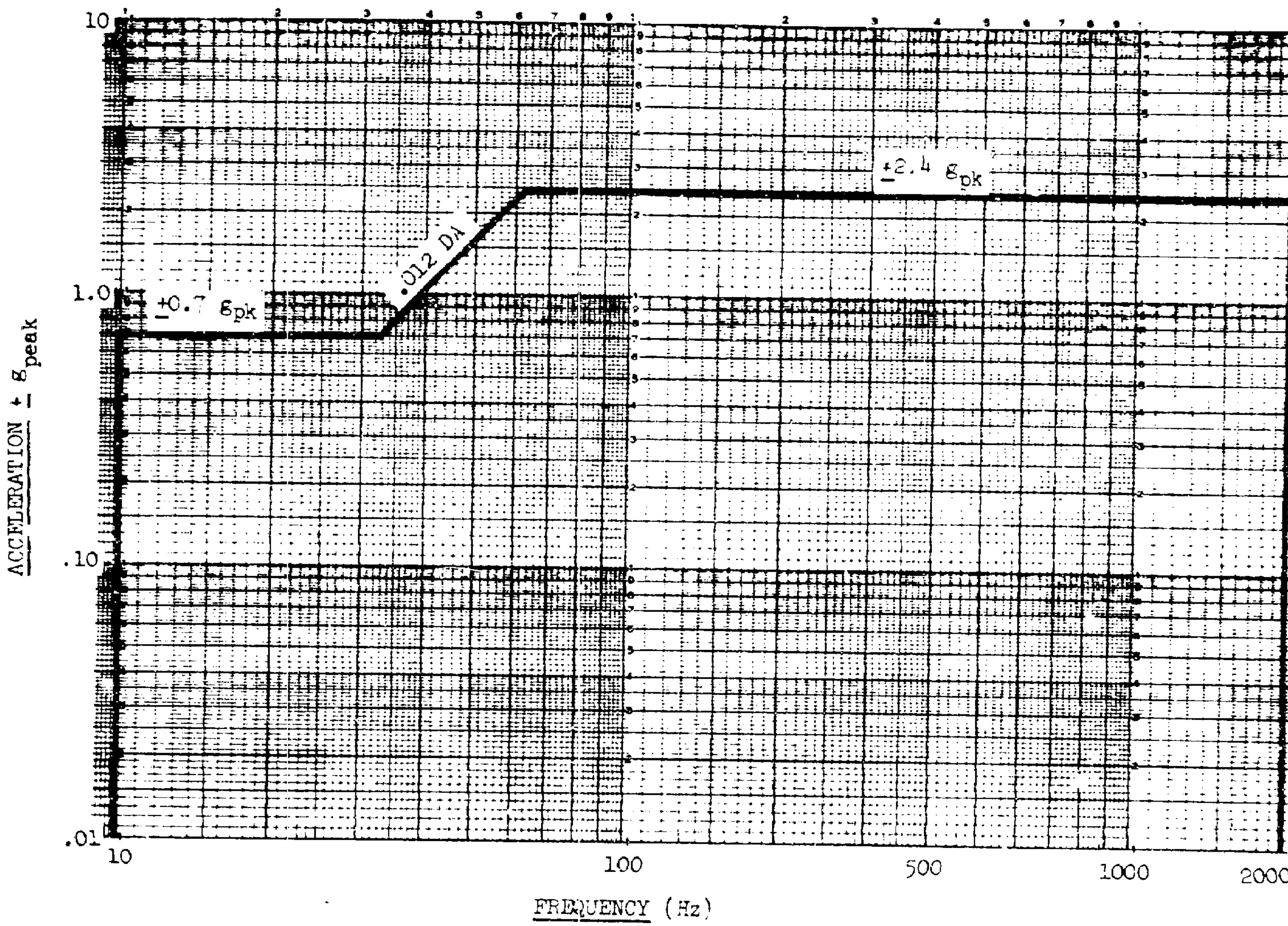


FIGURE 44 RECOMMENDED VIBRATION TEST ENVELOPE -- TURBOPROP INSTALLATIONS

TABLE 26 RECOMMENDED VIBRATION TEST PARAMETERS FOR INTERNALLY MOUNTED WRA's

PROPULSION TYPE	WRA LOCATION	VIBRATION TYPE	FREQUENCY RANGE (HZ)	APPLICABLE TEST ENVELOPE	DURATION AT MAX. LEVEL PER FLIGHT SEQUENCE	DISTRIBUTION OF TIME AT MAX. LEVELS*
Jet (fuselage mounted)	Forward of Engine Compartment	Random	10-2000	Fig. 41	Total Duration	Not Applicable
	Engine Compartment	Random	10-2000	Fig. 42	25 minutes	10 min at start of "take-off" analog 15 min at "dynamic maneuvering" analog
	Aft of Engine Compartment	Random	10-2000	Fig. 43	25 minutes	① 10 min of start of "take-off" analog 15 min at "dynamic maneuvering" analog
Jet (wing or pod mounted)	All	Random	10-2000	Fig. 41	Total Duration	Not Applicable
Turboprop	All	Sine Sweep	10-2000-10 @ 1.22 octaves/min	Fig. 44	25 minutes	12.5 min at start of "take-off" analog 12.5 min at "dynamic maneuvering" analog

* Test is to be conducted at 50% of maximum levels throughout remainder of required vibration test duration for each flight sequence.

① Two minutes at ground level and eight minutes at flight level

- As indicated in Section III, low pressure has a minimal effect on cooling capability.
- Low pressure, as an independent environment, has not been shown to contribute significantly to equipment failure.
- Inclusion of low pressure within the basic thermal test cycle would substantially effect the test time line and the ability to produce rapid temperature variations.
- Pressure testing would be hampered by the presence of humidity in that the presence of any residual water vapor interferes with chamber execution.
- In view of the relatively small contribution of altitude to WRA unreliability, the inclusion of a low pressure test outside the basic temperature cycle does not appear economically justifiable.

6.5 RELIABILITY DEMONSTRATION TEST PROFILE

The reliability demonstration test profile is the result of combining the developed thermal profile with the vibration profile.

A composite profile showing the interrelationships and proper phase sequencing is presented in Figure 45 to illustrate the process for one cycle. The test consists of repeated applications of this cycle until the accept/reject criteria are satisfied. It should be noted that the humidity exposure is an independent cycle, inserted periodically as shown in Figure 27.

6.6 SUBSYSTEM DEMONSTRATION TESTING

This study has addressed itself to analyzing the environment of each WRA. Consequently, the profiles presented herein are applicable to WRA's and single unit equipments. Although the study was WRA oriented, it is not suggested that reliability demonstration testing at the subsystem level be eliminated since, from both economic as well as schedule considerations, it is more desirable to test a group of WRA's than to test them individually. These profiles would be applicable to systems whose WRA constituents are the same MIL-class and are located in the same vibration zone. As system complexity increases, the likelihood of satisfying both the above conditions

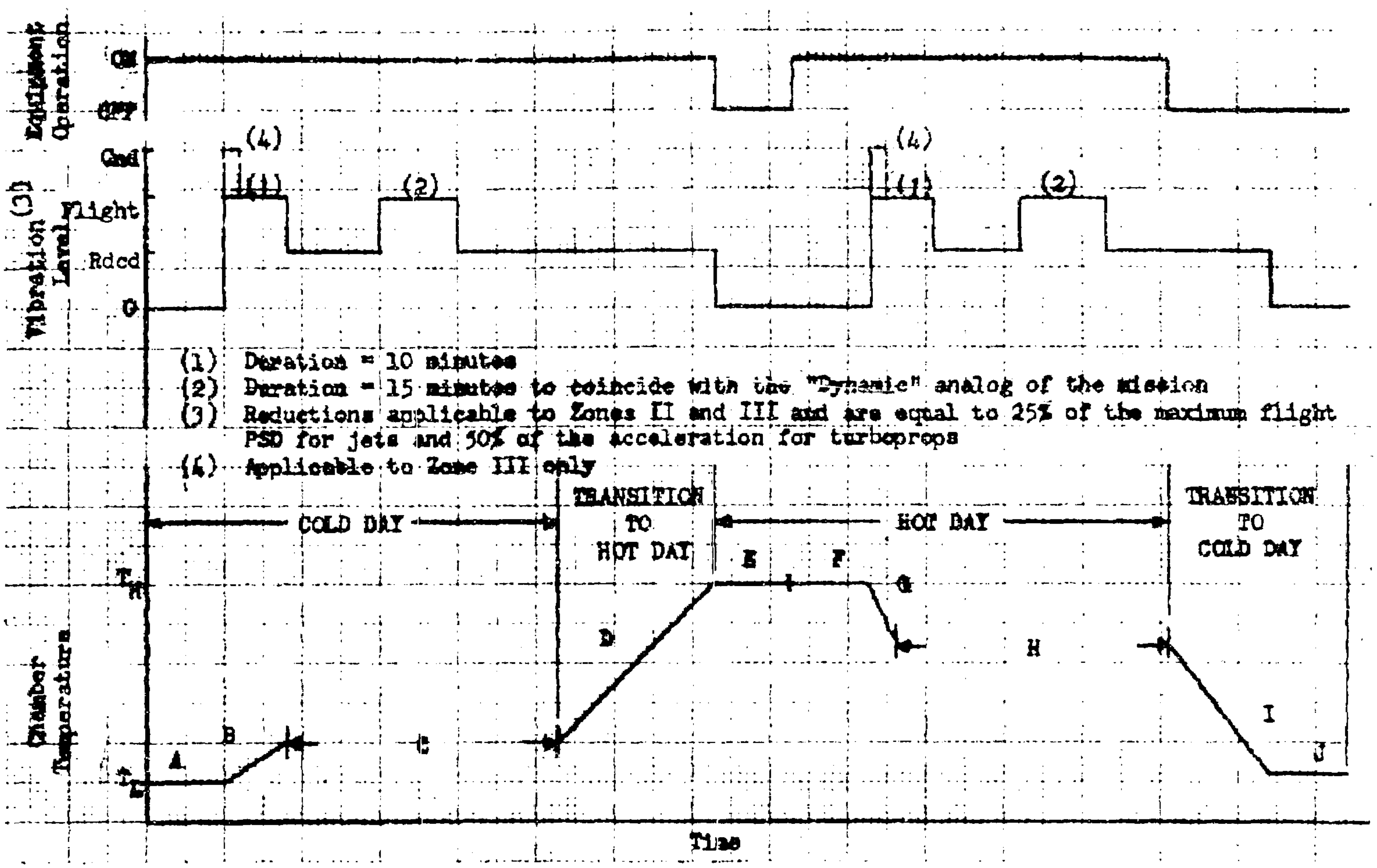


FIGURE 45 TYPICAL COMPOSITE DEMONSTRATION PROFILE

decreases, and thus the selection of proper test levels is not apparent. In the extreme case of a system composed of Class I and II WRA's located in all three vibration zones, six different test combinations would be required.

Several alternatives were investigated but no universal solution was found. The options investigated and the disadvantages of each are listed below:

Thermal

- utilize separate temperature-chambers for each MIL-Class (which implies separate vibration installations for each chamber) -- high cost
- provide a separate enclosure within a chamber to isolate the different MIL-Class WRA's -- additional test complexity and potential difficulties in controlling the environment in the enclosure
- test at a compromised level -- lose the association with the anticipated field environment

Vibration

- utilize a separate vibration exciter for the WRA's installed in each vibration zone (which implies separate chambers) -- high cost
- test at a compromised level -- lose the association with the anticipated field environment
- test at the highest level -- if failures occur on WRA's that should have been tested to the more benign levels, are the failures related to time or level?
- determine the test time reduction for Zones I and II WRA's if tested to the Zone III level (based on the accelerated test level theory, i.e., stress versus number of cycles relationship (S-N)). Perform the test on all items at the Zone III level, discounting any failures that result in the Zone I and II WRA's after the predetermined time is accomplished -- it is often not clear whether the failures encountered after the accelerated level testing is completed are related to vibration or thermal exposure

- same as above except, remove that portion of the system equipment from the vibration test setup, and install the equipment in a separate non-vibrating thermal chamber after the equipment has reached the determined accelerated test level period -- the major disadvantage to this again is cost.

Trade-off analyses have to be performed to make the proper selection from among these alternates. The outcome of the analysis, however, is highly dependent on the specifics of the system being evaluated; viz., number of WRA's, their distribution among the different test levels, WRA criticality, predicted failure rates, etc. It was concluded that since each system has to be evauated independently, no generalizations can be made and the trade-off analysis would have to be performed by the procuring activity and contractor as each situation arises.

SECTION VII

PREDICTIONS

A secondary objective of the study was to identify those factors which tended to contribute to the difference between predicted and field reliability. The analysis was conducted in a manner similar to that performed for demonstrated reliability. Differences in reliability between the prediction and the field were compared with differences in each suspected factor to determine if any general pattern was evident.

The predictions originally performed by each equipment manufacturer utilized a wide variety of Government sources for failure rates (MIL-HDBK-217, MIL-HDBK-217A, FARADA, etc.), as well as failure rates based on the sellers' own field experience. Further, each manufacturer also tended to apply his own ground rules when making predictions. In order to establish a common baseline for each of the selected items, new predictions of the reliability for each WRA were performed using the same sources and ground rules for each. This approach provided a consistent comparison of predictions with reclassified demonstrated or field MTBF's. The selected technique utilized MIL-HDBK-217B Coordination Copy (ref. 23), since this represented the most current information available at the time and would probably provide closest correlation with field failure rates.

A standard set of ground rules and assumptions were established where reference 23 did not include failure rates for certain parts included in this study. It should be noted that these criteria were applied to all of the selected equipments. The following major ground rules and assumptions were made:

- In several instances (capacitors and high stress ratio diodes), values for certain parameters (stress ratio vs. temperature) were of a magnitude not available from the curves. The RADC Notebook (ref. 24) was used in these cases.
- For certain parts (e.g., mechanical items), MIL-HDBK-217, Appendix II was used.

- Seller failure rates were utilized in those few instances where a device was considered proprietary by a seller and no information was available.
- The application factor for Group I and II transistors was assumed to be linear unless the device was used as a logic switch.
- The voltage stress for Group I Transistors was established at 60% since precise stress values were not available.
- Insert material for connectors was assumed to be Type A unless otherwise stated or known (coax).
- Voltage stress ratios for Group IV diodes were defined as 100%.
- Unless otherwise known, filter configurations were assumed to be Pi.
- An application factor of 2.0 was assigned to "CY" capacitors having a capacitance greater than 10,000 PFD.
- Hybrid thin film equations were used to determine failure rates of resistor networks.

After the performance of this analysis, the final version of MIL-HDBK-217B (ref. 25) was released. A review of that document indicated significant differences between it and the coordination copy but, because of the constraints on time and effort, reprediction of the reliabilities using the released document was beyond the scope of the contract. Thus, the results of the analysis presented herein as based on the prior completed predictions.

The reclassified field MTBF's as determined in Section IV were the measures of field reliability used for this analysis. Table 3 of Appendix D, presents the repredicted and reclassified field MTBF's for each WRA. The laboratory demonstrated MTBF for each WRA are also included for comparative purposes.

The analysis consisted of determining the ratio between the repredicted and the field MTBF's and to determine how these calculated ratios were distributed among the categories of each design or end-use characteristic investigated. Two excessively large ratios (WRA's 92 and 94) were eliminated

from the analysis as being inconsistent with the rest of the data. The significant relationships between the average MTBF ratios and the factors investigated is presented in Table 27 and the conclusions that could be drawn, relative to prediction technique improvement, are presented below. No significant relationship was found for weight, packaging density, power density or percent microcircuits. Since the MTBF ratios are a function of a failure rate source document that is now suspect, use of the absolute value of these ratios is limited. If the changes between the coordination copy and the released version of MIL-HDBK-217B are essentially constant for each WRA, then the relative weight of each category of a factor is preserved.

Function: All the functions had approximately the same average ratio with the exception of Displays and Controls, Electromechanical Devices, and Racks and Cabinets. These latter three functions had an average ratio 3 to 4 times worse than the other functions suggesting the following:

- Displays and Controls: The prediction technique does not make provision for high rates of ON/OFF cycling, or periodic adjustments that are common for this group. Also, the potentiality for greater field abuse exists.
- Electromechanical: The predictability of failure rates for "low population-unique design" parts (gyros, gear trains, tape drives, etc.) that make up this group is always subject to possible error because of the limited data base upon which statistics can be calculated.
- Racks and Cabinets: Connector and harness failure rates should be reviewed.

Cooling Method/MIL-Class: No variation with MIL-Class was observed which indicates that the prediction technique accuracy is not affected by anticipated temperature altitude environment. Yet, supplementally cooled WRA's have closer correlation than the ambient cooled WRA's, in approximately a 5:9 relationship. This suggests that prediction techniques should not have environmental factors for "airborne inhabited" and "airborne uninhabited" but rather, "airborne supplementally cooled" and "airborne ambient cooled."

TABLE 27 AVERAGE PREDICTED/FIELD MTBF RATIOS FOR SELECTED WRA DESIGN AND END-USE CHARACTERISTICS

Function

Function	Average MTBF Ratio*
Receivers and Transmitters	3.4
Interface Equipment	3.5
Computers	3.5
Power Supply	3.7
Signal Processing	4.6
Displays and Controls	12.9
Electro-Mechanical	13.9
Racks and Cabinets	15.4

Cooling Method

Type of Cooling	Average MTBF Ratio*
Supplemental	4.9
Ambient	8.8

MIL Class

MIL Class	Average MTBF Ratio*
1A	6.4
1	6.1
2	7.4

Aircraft Propulsion

Type	Average MTBF Ratio*
Propeller	6.1
Jet	7.2

Mounting Method

Type of Mounting	Average MTBF Ratio*
Isolator	5.2
Hard	8.1

Number of Parts

Quantity	Average MTBF Ratio*
Under 500	11.0
500 to 1000	6.1
1000 to 2500	3.8
Over 2500	3.0

High Reliability Parts

% TX, ER or Better	Average MTBF Ratio*
Under 20	8.5
20 to 50	5.7
Over 50	4.8

*Predicted MTBF/Field MTBF

TABLE 27 AVERAGE PREDICTED/FIELD MTBF RATIOS FOR SELECTED WRA DESIGN AND END-USE CHARACTERISTICS (Continued)

Burn-In Hours (Production Units)

Hours	Average MTBF Ratio*
Under 100	8.1
100 to 200	5.8
200 to 500	5.4

Burn-In Hours (Production Units)/ Predicted MTBF

Interval	Average MTBF Ratio*
Under 0.001	47.2
0.001 to 0.01	9.3
0.01 to 0.1	5.6
0.1 to 1.0	2.8
Over 1.0	1.6

Burn-In Failures

Number of Failures	Average MTBF Ratio*
Under 0.5	8.9
0.5 to 1.0	4.5
1.0 to 4.0	4.4
Over 4.0	1.8

Burn-In Failures/Number of Parts

Interval	Average MTBF Ratio*
Under 0.001	4.8
0.001 to 0.01	10.2
0.01 to 0.10	33.3

Burn-In Failures/ Expected Failures**

Interval	Average MTBF Ratio*
Under 0.5	6.0
0.5 to 2.0	1.8
Over 2.0	7.9

*Predicted MTBF/Field MTBF

**Burn-In Hours/Predicted MTBF

Propulsion Type/Mounting Method: Approximately the same lack of MTBF agreement exists regardless of the propulsion type. An 8 to 5 difference was observed between hard and isolator mounted WRA's. This suggests the need for a "mounting" factor in the prediction.

Number of Parts: The correlation improves as the quantity of parts increases. This just suggests that any inaccuracies in part failure rate data get averaged out when the part population is large. It further points out the pitfalls in making a prediction on a WRA that has a rather small parts population.

Percent High Reliability (TX or ER) Parts: The greater the percentage of MIL or lower quality grade parts, the worse the correlation between predicted and field MTBF's suggesting that the "quality factors" require expansion over a broader range.

Burn-In: If one again (see para. 5.2.3) assumes that burn-in failures are proportional to the emphasis on quality issues during manufacture and are also a barometer of the effectiveness of the burn-in test, the data indicates that the greater the emphasis on these factors the better the correlation between predicted and field MTBF's. Further, the data also indicates the longer the WRA burn-in test duration the better the correlation, as would be expected. The reference 23 prediction technique requires that the "quality factor" used to adjust the base failure rate be determined by jointly considering part quality and WRA manufacturing quality. Perhaps some inconsistency is introduced by this approach since it is not clear which of the following two situations has the higher reliability:

- high quality parts with poor manufacturing quality or
- low quality parts with high manufacturing quality.

SECTION VIII

CONCLUSIONS AND RECOMMENDATIONS

8.1 CONCLUSIONS

This study of 95 distinct WRA's representing a cross section of avionic types and applications, indicated that differences between laboratory demonstrated and field observed reliability is attributable to a wide variety of factors. Both differences in data, end-use application, and environments between the laboratory and the field were found to relate to reliability differences.

- Inconsistent ground rules and failure scoring criteria account for a significant portion of the difference between demonstrated and field MTBF's. Anomolies which should be considered countable are often excluded by demonstration test ground rules. Current military data reporting systems lack detail in describing malfunctions, resulting in counting incidents which should be excluded.
  - Difficulties in collecting WRA operating time data results in flight time being used as the time base for field MTBF determination. Demonstration tests use operating hours as the time base.
  - Reclassification of failures (field and laboratory) in accordance with study developed criteria and use of estimated field operating time resulted in, on average:
    - o 30% to 40% decrease in demonstrated value
    - o 100% increase in field value
- A substantial difference in reliability remained after the reclassification. Fifty percent of the WRA's studied had field values three times lower than its corresponding demonstrated value. The average ratio between demonstrated and field MTBF's was approximately 8.0.
- WRA's which historically have had more design effort in minimizing field environmental effects (e.g., RF equipment, high power density) had better reliability agreement.

- The reliability agreement between the laboratory and the field was poorer on those items that had a potentiality for abuse in the field (e.g., high packaging density, displays and controls).
- WRA's having relatively large microcircuit or "Hi Rel" part complements had better demonstrated to field reliability agreement, thus indicating the benefits to be gained by using these parts.
- Those items that had the more effective burn-in testing on production units tended to have the better reliability agreement. This indicates the necessity for adequately specifying a production unit burn-in both in duration and in environmental exposure.
- Relationships were found between reliability differences and several temperature related measures for ambient cooled WRA's, including:
  - minimum ambient temperature
  - operating time at low temperature
  - maximum ambient temperature
  - temperature rate of change

indicating that the current MIL-STD-781 tests of only requiring dwells at the temperature extremes with moderate rates of change between the limits is not an adequate test. No provisions exist for evaluating the item under conditions of rapid and frequent temperature cycling.

- For forced air cooled WRA's, the significant temperature related measures included:
  - maximum ambient temperature
  - maximum cooling air temperature
  - minimum ambient temperature
  - maximum cooling air flow rates
  - operating time at low temperature

indicating that the effects of field typical cooling air parameter variations are never evaluated during the demonstration test. Furthermore, the effect of this lack of variation in the laboratory is to essentially shield the article from any temperature effects.

- The significant relationships between vibration measures and reliability differences included:
  - level
  - duration

indicating that the MIL-STD-781 vibration test of requiring 10 minutes of sinusoidal vibration each hour at one non-resonant frequency between 20-60 Hz is not representative of the field environment. The test article is never exposed to those frequencies occurring in the field, that produce failures. The vibration test duration was found to be a poor representation of the accummulated field vibration time. The lack of reliability agreement was more pronounced in WRA's installed in jet aircraft where the field environment is random.

- Moisture, which data and previous experience indicates is a major source of field failure, is not a test requirement of MIL-STD-781.
- No evidence could be found that altitude or voltage cycling, as independent environments, significantly contribute to field problems.
- Although the analysis of differences between predicted and field reliability was terminated when the released version of MIL-HDBK-271B was issued, some preliminary observations were made:
  - reliability differences were better correlated with cooling method than MIL-Class
  - hard mounted WRA's had poorer agreement than vibration isolated items
  - displays and controls, as a functional group, had poor agreement (ON/OFF cycling not properly accounted for in prediction technique)
  - WRA's with low parts population or specialized unique design components had poor agreement (small data base)
  - WRA's with a proportionately large quantity of Hi Rel parts had the better correlation (suspect "quality factors")
  - the more effective WRA burn-in test, as measured by duration and number of failures detected, resulted in closer prediction to field reliability agreement

8.2 RECOMMENDATIONS

The results and conclusions of the study suggest several general areas of improvement for assuring future closer agreement between demonstrated and field reliability. These include:

- Revision of the demonstration test temperature profile and method of profile construction as described in paragraph 6.2. This would feature:
  - mission profile orientation
  - variations in chamber temperatures as a function of changes in flight conditions and thus approximating compartment temperatures and exposing the item to more temperature cycling
  - periodic dwells at temperature extremes to simulate non-operating conditions
  - maximum permissable variation in cooling air temperature and flow rates
  - coupling of cooling air variations with chamber temperature variations
  - temperature rates of change equal to flight levels
  - equipment ON/OFF schedule designed to assure that internal components are properly exposed to temperature variations
- Revision of the demonstration test vibration profiles as described in paragraph 6.3. This would feature:
  - random vibration for items installed in jet aircraft
  - sinusoidal sweeps for items installed in propeller driven aircraft
  - exposure to all input frequencies up to 2000 Hz
  - levels to approximate those experienced in the field
  - increased duration to more adequately reflect accummulated flight time
- Inclusion of a humidity test, as described in paragraph 6.2, to periodically evaluate the effects of moisture during the demonstrtion.

- Incorporation of precisely defined failure criteria and scoring ground rules as described in paragraph 4.2.2. These should clearly describe those anomolies to be considered relevent and non-relevent as well as provide the minimum conditions for reclassifying a relevent failure as non-relevent. A strong recommendation is made that procuring activities critically review and evaluate failure exclusion groundrules in accordance with the ultimate application of the hardware in the field.

- Experience and the study data have indicated the positive benefits derived from requiring a strong end-item burn-in test. This test should be required of all future procurements and include:

  - adequate environmental exposure (temperature and vibration)
  - sufficient duration to screen out most workmanship failures

- The prediction technique should be reviewed with the objective towards adding or changing some of the modifying factors. These include:

  - replace "habitation" factor with "cooling method" factor
  - include a "mounting method" factor
  - review numerical relationship among the quality grades to modify the "quality" factor
  - include an "end-item quality assurance" factor

8.3 AREAS FOR FUTURE INVESTIGATION

- The temperature profiles developed herein are applicable to MIL-E-5400 Class I and Class II equipments. Although these are the most prevalent classes of avionic equipment, temperature profiles for the applications not covered in the study should be developed for completeness.

- The present MIL-STD-781 test procedure requires vibration testing in a single axis. Generally, the vertical axis has been the one used; yet, this is not necessarily the most critical one. Three axis testing might be preferred since the unit potentially experiences vibration in all three axes in the field. However, this would

result in additional cost and testing complexity. A study should be undertaken to investigate: the necessity for multiaxis testing, rules for determining the best single axis test, and the feasibility of "resultant axis" testing. In resultant axis testing the equipment is oriented such that when vibration is applied along the input axis the equipment experiences simultaneous excitations in three mutually perpendicular planes.

- The vibration profiles developed in this study are only applicable to internally mounted equipment. No profiles were developed for external or surface mounted equipment since it was felt that each case had to be evaluated separately. An investigation should be conducted to determine if sufficient generalizations in terms of engine characteristics, profiles, and aircraft structure can be made to develop a set of vibration profile recommendations for these items.

- The approach to development of the profiles presupposes that the intended end-use application is known. In order to use these profiles, one has to know: the aircraft, mission profile, location of the unit, etc. Profile recommendations need to be developed for those items where this information is not known or the unit is intended for several aircraft applications (e.g., GFE).

- A test program should be conducted on a representative sample to demonstrate the effectiveness of the recommended profiles.

- A draft revision to MIL-STD-781 which incorporates the study results should be written.

- This study addressed itself to testing WRA's or single unit systems. It was recommended that system level testing be continued, as in the past, on those equipments containing more than one WRA. Although testing alternatives were presented, no guidance could be offered for those situations where each WRA requires a different test level. It was argued that each such situation should be considered on its own and its alternate solution would depend on the trade-offs among

cost, schedules, number of different test combinations, criticality, etc. It is recommended that a study be performed to identify the elements to be considered, the relationships and the logic flow for such trade-off analyses.

- The study identified the need for including a "quality assurance" factor in the prediction technique. This would be independent of the "parts quality" factor and would represent the emphasis placed on quality issues during end-item assembly and test. A study should be conducted to develop the approach for determining, and specific values of, this factor.
- This study indicated the beneficial results associated with a strong end-item burn-in test. In addition, the data suggested a method by which the duration of such a test could be specified. A study to establish the methodology for determining burn-in requirements, i.e., environments, criteria, and duration, should be undertaken.
- The study identified the difficulties associated with establishing demonstration test requirements on a subsystem level where WRA's are subject to different environments. Since the successful completion of an environmental qualification test is a precursor to the demonstration test, the results of the qualification could conceivably be used as an aid in determining demonstration levels. A study should be conducted to determine how one might capitalize on qualification results and/or modify qualification tests for special situations to complement the demonstration.
- This study recommends the inclusion of a humidity exposure. Current laboratory practices prohibit the introduction of corrosive and/or conductive contaminants. This method does not produce an exposure which is completely analogous to that experienced in the field since it precludes the introduction of dissolvable minerals. It is recommended that a study be performed to develop a procedure to include the introduction of field representative quantities and distribution of dissolvable minerals prior to or during the humidity exposure.

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APPENDIX A

WRA Descriptions

The following paragraphs provide a brief description of each WRA included in the study. Information regarding the function, configuration, construction, cooling method, mounting method, and location is generally provided to give the reader an insight into the type and variety of WRA's selected. The WRA's are presented by major functional grouping, i.e., data processing, RF receiving and transmission, etc.

RF R/T - WRA's which either transmit or receive RF signals are included in this grouping.

WRA No. 1 is an RF receiver and is located in the upper left-hand shelf of the aft equipment bay structure. Two bolts in the front and two spring loaded alignment pins in the rear provide a rigid mounting for the assembly. All fuses, connectors and an elapsed-time meter are on the front panel. During normal operation, cooling air is scooped into the rear air inlet and fed out the front exhaust to maintain a proper operating temperature.

WRA No. 2 is an RF transmitter consisting of 19 oil-cooled ceramic tetrodes operated as a class A distribution amplifier. They are divided into driver, intermediate and final stages which amplify the output of the control to the proper level for transmission. There are four basic modes of operation and, depending upon the mode selected, various sub-modes and routines.

WRA No. 3 is an RF receiver. The unit is functionally subdivided into three parts: transmitting, search receiving, and terrain clearance receiving. The unit is located in the nose of the fuselage and is accessible by raising the fiber glass radome. It is ambient cooled and hard mounted.

WRA No. 4 is an RF receiver and is housed in the upper right portion of the aft equipment bay structure. Two bolts in the front and two spring-loaded alignment pins on the rear provide a rigid mounting for the assembly. All fuses, connectors, and an elapsed time meter are on the front panel. During normal operation, cooling air is scooped into the rear air inlet and is fed out the front exhaust to maintain a proper operating temperature.

WRA No. 5 is an RF receiver and is located in the left wing fillet. The receiver has a metal case with three electrical and six coaxial connectors. An elapsed-time indicator is visible on the front of the receiver unit. It is hard mounted and ambient cooled.

WRA No. 6 is an RF receiver and is housed in the forward portion of the receiver compartment. Four bolts in the front and two spring-loaded alignment pins in the rear provide a rigid mounting for the assembly.

All fuses, connectors, and an elapsed-time meter are on the front panel. During normal operation, cooling air is scooped into the rear air inlet and is fed out the front exhaust to maintain a proper operating temperature.

WRA No. 7 is an RF transmitter located in the fuselage nose, which provides continuously adjustable high-energy pulses of selectable width and repetition rate. The unit contains 13 removable assemblies and three harness assemblies with integral filters, connectors, and relay circuitry. It consists of two separate cast aluminum rectangular housings secured together to provide a single unit, and four shock mounts provide for installation in the aircraft. Forced air is required for proper cooling. The forced air is applied to an intake opening in the bottom of the housing, circulated past four heat-exchanger plates, and exhausted through vents on the top of the housing. Nitrogen or dry air is required for proper pressurization.

WRA No. 8 is an RF transmitter consisting of 19 oil-cooled ceramic tetrodes operated as a class A distribution amplifier. They are divided into driver, intermediate and final stages which amplify the output of the control to the proper level for transmission. There are four basic modes of operation and depending upon the mode selected, various sub-modes and routines.

WRA No. 9 is an RF receiver-transmitter, located in an equipment bay, that is capable of receiving and transmitting voice and data. The unit is housed in a 1/2-ATR case with the right side cover removable for module accessibility. All modules interconnect through a printed circuit side-board which also contains a terminal field for soft wire interconnection to the I/O connector. The receiver-transmitter contains mechanical filters, crystal filters, and wide dynamic range front-end circuits that provide rejection of strong adjacent channel signals.

WRA No. 10 is an RF receiver and is housed in the forward portion of the receiver compartment. Four bolts in the front and two spring-loaded alignment pins in the rear provide a rigid mounting for the assembly. All fuses, connectors, and an elapsed time meter are on the front panel. During

normal operation, cooling air is scooped into the rear air inlet and is fed out the front exhaust to maintain a proper operating temperature.

Signal Processing - Items which deal directly with electronic signals, i.e., processing, modulation, amplification, attenuation or filtering, comprise this group.

WRA's Nos. 11 and 12 are signal processing units that decode firing signals. Each unit contains two printed circuit cards hard wired to each other and an interface connector. Most of these units are hard mounted to a weapons rail.

WRA No. 13 is a signal data converter which provides timing pulses. In addition, the unit processes the RF returns for presentation by various displays. It is housed in an aluminum case with four mounting brackets and a carrying handle. Six electrical connectors and an elapsed-time meter are on a connector panel at one end of the unit. The unit is located in the nose.

WRA No. 14 is a signal processor which generates pulsed outputs in response to an input. The unit consists of a single equipment cabinet, hard mounted in a frame enclosure in the fuselage equipment bay. Cooling is provided by controlled forced air from a vapor cycle system. Various connectors, controls and an elapsed time meter are located on the cabinet's front panel.

WRA No. 15 combines RF inputs from various units into one signal representing the sum of the inputs, and applies this combined signal for further processing. The unit is hard mounted on the fuselage top deck and is ambient cooled.

WRA No. 16 is a network which suppresses transients in the 115 vac and 28 vdc aircraft power lines. The network is mounted in the nose of the aircraft.

WRA No. 17 is a comparator-converter which receives and processes video signals. It is housed in the lower right-hand corner of the aft equipment bay structure. The assembly is secured in place by two spring-loaded alignment pins at the rear and by two bolts in the front which

attach to the aft equipment bay structure. An elapsed time meter, cable connector, and fuses are mounted on the front panel of the assembly. During normal operation, cooling air is scooped into the rear air inlet and is fed out the front exhaust to maintain a proper operating temperature.

WRA No. 18 is a unit that splits a combined signal sample into six signals of equal magnitude and applies these signals to various receivers. The unit is mounted in the tail fin area of the aircraft and is ambient cooled.

WRA No. 19 is a control unit used to provide an RF drive corresponding to the assigned frequency of either of two transmitters. An RF sample of the carrier frequency produced is sent to other subsystems for sample display. A BIT feature is included to self test the unit. The unit is cooled by liquid circulated within a heat exchanger. The exchanger is cooled by external air.

WRA No. 20 is a signal processor containing receiver, gate and logic channels, a BITE network and a power supply. The unit is hard mounted in the right forward equipment bay and is cooled by an internal fan.

WRA No. 21 is a 3-pole bandpass filter that is tunable in four bands. The filter provides front-end protection to the receiver-transmitter from strong off-frequency signals and also provides selectivity for the receiver-transmitter. It is housed in a 1/4-ATR (short) case and contains four plug-in printed circuit card assemblies. The unit is completely solid state and no special cooling is required. It is located in an equipment bay.

WRA No. 22 is a broad band filter assembly and is located within the aircraft wing. It is housed in a metal case with two electrical connectors and six coaxial connectors. It is hard mounted and ambient cooled.

<u>Interfaces</u> - Devices which act as interfaces, junction boxes, couplers and converters make up this category.

WRA No. 23 is a display/converter which functions as the interface between a computer and indicator and display units. It is forced air cooled and isolator mounted. It is located in the fuselage.

WRA No. 24 is an analog-digital converter and serves as the interface between a computer and analog data devices. The unit contains 39 plug-in printed circuit cards, which are held in place by tie-down bars. To enable cooling of electronic components, large areas of copper extend outward from the plug-in printed circuit cards. Heat transfers from the components to the copper pad and ultimately to the chassis walls. The front panel of the unit contains controls, indicators, and seven operational connectors. It is located in the fuselage behind the cockpit.

WRA No. 25 is an interface box which provides for common distribution and preprocessing of signals for various displays and controls. Lamp-driver circuits provide for illumination of legend indicators on the BIT control, and dc outputs are provided for assemblies in the cockpit. This unit is located in the cockpit and is secured in place by two bolts which pass through a mounting structure at the front and two holes at the rear which mate with tapered locating pins.

WRA No. 26 is a converter which functions as the interface unit for control, data transmission, data storage, and navigation parameter display between a computer and navigation equipment. The unit is isolator mounted and receives supplemental cooling air. It is located in the fuselage equipment bay.

WRA No. 27 is a control interface unit and is part of a computer set. It provides the controls, displays and circuitry required to enter and transfer data, control computer operating modes, and control radar cursors. The unit consists of three removable subassemblies. Forced air cooling is supplied through a vertical air inlet manifold. It is located in the cockpit.

WRA No. 28 is an interface unit which provides an interface between a computer and the aircraft navigation system. The computer interface is located in the aft equipment bay structure. It is secured in place by two drilled mounting plates at the front, and two alignment pin sockets at the rear. Signal connectors, power connectors, fuses, and an elapsed time meter are located on the front panel, and a test connector is located at

the rear. Cooling air is circulated through the assembly via four inlet ports at the rear and four exhaust ports at the front.

WRA No. 29 is an interface unit which provides the capability of communications between digital data equipment over a radio link. The unit converts binary information to a phase-encoded audio format suitable for hf or uhf radio transmission and vice versa. A card cage within the case supports up to 30 perpendicularly mounted plug-in circuit cards. Cooling of the data terminal is accomplished using forced-air cold-plate techniques. It is located in the equipment bay.

WRA No. 30 is an interface unit which provides computer data for aircraft radar operation. The equipment receives selected video and all required range and azimuth timing signals to digitally process the video into computer data. The unit is housed in a rectangular aluminum case that is locked in place by two latch sets. Cold plate heat exchangers are utilized with forced-air cooling to satisfy the cooling requirements. The unit is located in the aircraft nose.

WRA No. 31 is an interface box which provides distribution and preprocessing of signals for video displays and for audio signals. This unit is located in the aft cockpit and it is secured in place by two bolts which pass through a mounting structure at the front and two holes at the rear which mate with tapered locating pins. Except for the elapsed time meter at the front of the assembly, there are no controls or indicators. External connectors are located at the front top.

WRA No. 32 is a converter which acts in an interface capacity between a computer and data links. It is located in the fuselage equipment bay and requires forced cooling air.

WRA No. 33 is a high-speed, automatically tuned hf antenna coupler that transforms the complex impedance of the antennas to a value that is suitable as a load for a power amplifier. The unit is housed in a 3/4-ATR case and contains 12 removable modules. The modules are printed circuit cards and modularized assemblies. The WRA is forced-air ambient cooled by a blower and is located in an equipment bay.

WRA No. 34 is an interface unit which provides signal data interface control within a computer group between the processor, computer control, and tape recorder. It is forced air cooled and isolator mounted within the fuselage equipment bay.

WRA No. 35 is an interface unit which accepts synchro signals from various electromechanical sensors and supplies compatible synchro outputs to other requiring systems. The WRA consists of three modules and a self test assembly contained within a frame. The unit is hard mounted in the aft equipment bay and is ambient cooled.

WRA No. 36 is an antenna interface unit which generates the appropriate interlocks required for system and antenna protection. The unit is housed in a 1/4-ATR case and contains four plug-in printed circuit card assemblies. The unit is isolator mounted and requires no supplemental cooling air. It is mounted in the equipment bay.

Data Processing - This category includes items which perform computational (arithmetic) and similar functions.

WRA No. 37 is a general-purpose digital computer, which processes real-time control applications. In these applications, bombing and navigational computations are made based upon stored and computed flight data. Comprehensive self-test features are built into the computer to assist in fault isolation. The exterior structure provides cooling, interface connection, and electromagnetic shielding. The unit is mounted in the cockpit.

WRA's No. 38, 39, and 40 are computers which generate roll, pitch, and yaw control surface commands respectively. Each unit is housed in a structural box assembly and the two main structural members provide the mounting for all circuit board connectors. Electronic components within each computer are mounted on circuit boards accessible through the top cover. They are all located in the equipment bays.

WRA's No. 41, 42, and 43 are three types of arithmetic and control assemblies that act collectively as a central processor unit and perform five control functions: instruction, arithmetic, memory, program level and

input/output. They are forced air cooled and isolator mounted and are located in the fuselage.

WRA No. 44 is a computer consisting of various generator and detector circuits, BITE circuits and a power supply. The unit is hard mounted within the wing and is cooled by an internal fan. Ten electrical connectors, one cooling air intake screen, an elapsed time indicator and an overheat indicator - reset button are on the unit's front panel.

WRA No. 45 is a computer which generates coded signals in response to specific inputs. It is housed in a cabinet which is hard mounted to a frame type enclosure in the fuselage equipment bay. Forced air cooling is provided by controlled air from a vapor cycle system. Various connectors, controls and an elapsed time meter are located on the front panel of the cabinet.

WRA No. 46 is a navigation computer that interfaces between various navigation subsystems and auxiliary equipment. It consists of an analog-to-digital-to-analog converter and a miniature general-purpose computer. The unit is located in an unpressurized equipment bay and is forced air cooled.

WRA No. 47 is a core memory assembly which consists of a destructive readout, coincident current, core stack assembly and a memory selector. The memory selectors contain the necessary circuits for addressing the memories and for providing access by two central processors on a time-sharing basis. The unit is forced air cooled and isolator mounted. It is located in an equipment bay.

WRA No. 48 is a processor which processes data inputs from other aircraft systems for display. Based on the mode selected and navigation submode selected the processor sets the appropriate priority for each indicator and generates deflection signals necessary for display of required data. The unit is forced air cooled and isolator mounted and is located in an equipment bay.

WRA No. 49 is a computer containing a power supply and a memory which operates in various modes. The unit is housed in a single cabinet which is

hard mounted to a frame enclosure located in the fuselage equipment bay. Cooling is provided by forced air from the aircraft's vapor cycle system. A temperature sensing switch is located on the top panel of the cabinet.

WRA No. 50 is a combined interface and processor unit which together form a stored program, parallel, binary computer whose purpose is to receive data inputs, process these inputs with programmed routines and provide capability for display and return of processed data. It is located in the aft equipment bay structure and is forced air cooled.

WRA No. 51 is an air data computer which computes true air speed, impact pressure and altitude from static and total pressure supplied by the pitot static system. These computed quantities are supplied upon request to various components. The unit is fully automatic and is completely solid state. It consists of 16 printed circuit cards which plug into a mother board mounted to the chassis.

WRA No. 52 is a 4K coincident-current, random access-type memory whose function is to refresh the symbols for three independently operated indicators. It is located in the fuselage equipment bay and requires forced air cooling.

Indicators and Controls - Video display, control and personnel indication functions located in the aircrew compartment are included in this category.

WRA No. 53 is a flight control panel consisting of switches and circuitry which permit engagement of stability augmentation or autopilot flight modes. The unit consists of a flat, machined aluminum plate to which the connector bracket, switches and electroluminescent panel are attached. The case is an aluminum can which fastens to the panel with four screws.

WRA No. 54 is an indicator which provides alphanumeric and indicator light presentations. The unit is located in the cockpit at the operator console, and is supported by two mounting pins at the rear and eight quarter-turn fasteners on the front panel. All indicators are located on the front panel and all connectors and elapsed time meter are located at the rear.

WRA No. 55 is an indicator which provides computer readouts and controls. To perform its various functions, this unit provides manual display mode control of computer data, display of system advisory flight and navigation data, and manual control of magnetic variation and display of magnetic variation data.

WRA No. 56 is an equipment which provides the operator with controls necessary to apply power to various assemblies and to select modes of operation. The control is located in the cockpit at the operator console. The assembly is secured in place by four quarter-turn fasteners on the front panel. All operating controls are mounted on the front panel and all external connectors are located at the rear.

WRA No. 57 is a stick grip assembly which provides for control of the flight surfaces. It consists of a grip assembly, strain gage and connector assembly, electronic assembly and housing assembly. The electronic assembly consists of four amplifiers soldered to a flexible printed tape which terminates at a terminal board. The housing assembly supports an emergency disengage lever and switches.

WRA No. 58 is a unit which provides the operator with the controls necessary for operating various RF receivers. The control is located at the operator console, and is secured in place by five quarter-turn fasteners on the front panel. All operating controls are located on the front panel and all connectors are located at the rear.

WRA No. 59 is a control panel for an armament system. The panel contains the controls and indicators required to monitor and control the selection of stores, attack modes, and release modes. Lifting handles are provided on the front face, top and rear. Operating controls are on the front panel; a total time meter is provided on the rear face of the panel.

WRA No. 60 is a control which provides the operators with the means necessary to operate various RF receivers. The unit is located at the operator console, is secured in place by six quarter-turn fasteners on the front panel. All operating controls are mounted on the front panel and all external connectors are at the rear.

WRA No. 61 is a horizontal situation display. The display consists of a CRT providing an approximately five-inch in diameter display format. It provides a horizontal PPI or horizontal plan display as well as line written symbols. The unit is fan cooled.

WRA No. 62 is a BIT Control which provides the operator with the controls and indicators necessary to initiate and monitor the system built-in-test sequence. The sequence of BIT testing allows the operator to isolate a malfunction to an assembly.

WRA No. 63 is a control converter and is utilized in the system for control configuration of a data terminal set. The unit is comprised to two plug-in/bolt-in printed circuit assemblies and the main chassis. Access to the internal circuitry of the control is provided by means of a removable dust cover.

WRA No. 64 is an indicator which provides the operator with an indication of current system status. It contains 10 legend indicators. The unit is located on the operator's console, and is secured in place by four quarter-turn fasteners on the front panel. All operating indicators are located on the front panel, and all connectors are located at the rear.

WRA No. 65 is a display which presents navigational and other data to the flight crew. The unit contains a CRT that provides a five inch diameter display. The unit is cooled by an internal fan.

WRA No. 66 is a control/display which provides visual and audible signals. It is hard mounted in the cockpit and is cooled by an internal fan. The WRA is provided with an alphanumeric display which indicates the operational status of other system components. Three electrical connectors and an elapsed time indicator are accessible at the rear of the unit.

WRA No. 67 is a display which provides a video presentation of various signals. The unit is located in the cockpit and is mounted vertically on its left side and consists of a truss grid type of construction upon which the components are mounted. The display tube is installed in the center of the front panel and all operating controls are mounted on the front cover. The unit requires no supplemental cooling.

WRA No. 68 is a control set and contains all controls for operation of a subsystem. The unit is housed in a rectangular, frame-mountable aluminum case. Eight mounting screw holes, located along the top and bottom of the front panel, are provided for securing the unit to the main frame. A vane-axial cooling fan inside the case exhausts air through a ventilation hole in the case bottom when the unit is operating.

WRA No. 69 is a dual control unit for two channels of a communication set. The unit is comprised of five printed circuit plug-in assemblies, two of which are identical, and the chassis. The main chassis contains a printed circuit sideboard for all of the plug-in assemblies and the high dissipation elements of the power supply. The front panel of the WRA contains all the controls for both channels of the system.

WRA No. 70 is a computer control-indicator which serves as an input/output device for the computer. The unit is located at the operator console and is secured in place by eight quarter-turn fasteners on the front panel and two mounting pins at the rear. All controls and indicators are located on the front panel, and all external connectors and an elapsed time meter are located at the rear.

WRA No. 71 is a cockpit mounted control unit consisting of a front panel with aviation red lighting, various manually operated controls, and a logic card chassis. The chassis houses seven printed circuit cards which encompasses all of the low power logic functions. At the rear of the chassis is a separate enclosure where one power and three signal interface connectors are mounted along with the elapsed time meter. Two frequency generator modules are mounted to the removable left plate of the chassis.

WRA No. 72 is a control which provides the operator the ability to insert commands into a computer. Signals, generated within the control, command the computer to perform various functions. The unit is located in the cockpit at the operator console. It is secured in place by eight quarter-turn fasteners on the front panel. All controls and indicators are mounted on the front panel, and all external connectors are located at the rear. The unit is ambient cooled.

WRA No. 73 is a radar display which received video information, and processes these inputs to display the selected modes on a direct view storage tube. It also positions the true heading, command heading, range, and time indicators in response to control signals from a computer and a navigation subsystem. The unit is housed in a rectangular aluminum case mounted on a panel in front of the pilot. An elapsed time meter is on the right-hand side of the component. Guide rails along the top rear surface support the component and guide it into place where it is held by three screws that pass through holes in the indicator panel.

WRA No. 74 is a display providing video presentations of various signals. It is located in the cockpit and is supported at the rear by two tapered mounting pins, and at the front by two bolts which pass through the mounting structure and the bottom of the chassis. All operating controls are located on two removable front covers and all connectors and an elapsed time meter are located at the rear of the assembly. It is ambient cooled and hard mounted.

WRA No. 75 is a display which provides the operator with visual indications of information gathered, computed, or processed by various subsystems of the aircraft. The unit is contained in a rectangular aluminum housing having top and bottom access covers. The top cover is fitted with a carrying handle and a cooling air exhaust port. The unit is ambient cooled by means of internal fan and is hard mounted.

WRA No. 76 is a computer control and with its switches, indicators and associated circuits is used to control and monitor system, processor and tape recorder functions. These functions include system reset processor selection, program loading and manual tape control positioning, navigation function control, testing and fault indication. Forced air is used to cool the unit and it is mounted on isolators.

WRA No. 77 is a control which provides the operator with a means of controlling the various equipment functions. Dc voltages are provided at the output of the assembly to implement the control functions, as well as to provide indicator lamp illumination power to the other assemblies. The control is located in the cockpit at the operator console and is secured in

place by four quarter-turn fasteners on the front panel. All operating controls are mounted on the front panel and all external connectors are at the rear. It is ambient air cooled.

Power Devices - Power supplies and power switching units comprise this category.

WRA No. 78 is an AC/DC converter power supply which provides unregulated DC to computer memory power supplies and to a control assembly for power failure detection. The unit is shock mounted and is cooled by forced air. It is located in the fuselage.

WRA No. 79 is a power supply containing six voltage regulators which operate off a common power transformer. These regulators supply regulated voltages to various assemblies. The assembly contains an automatic load sensor and protection from excessive voltage output variations. The output of each regulator is sampled by a BIT generator which provides an indication when a power failure occurs. It is isolator mounted and forced air cooled. It is mounted to the aft fuselage.

WRA No. 80 is a power switching unit and consists of a front panel face, a main power switch card chassis and a dual secondary power supply. The power switches control 28 VDC power to several aircraft equipments. The power supply is of modular construction and contains four (4) printed circuit cards and the main frame where a capacitor storage bank and large power dissipating elements are mounted. The unit is forced air cooled and is located in the fuselage above the wing.

WRA No. 81 is a power supply which provides regulated DC for Memory Core Modules. It is forced air cooled and isolator mounted and is located in the fuselage.

WRA No. 82 is a low-voltage power supply and supplies, rectifies, regulates, and controls low voltages for various components. The unit is housed in a rectangular aluminum case with four mounting brackets. Cold-plate heat exchangers are used with forced air cooling to satisfy the unit cooling requirements. The unit is mounted in the nose.

WRA No. 83 is a power supply which provides operating power to other assemblies in the cockpit and contains five voltage regulators which operate off a common power transformer. The power supply is located in the nose right equipment bay of the aircraft and is secured by sliding the assembly onto an isolating tray. Mounts, fastened to the bottom of the isolating tray, latch onto metal brackets on the assembly in order to secure it in place. It is forced air cooled.

WRA No. 84 is a linear power amplifier. The unit provides amplification, gain compensation, and signal limiting of the input RF signal. The WRA contains a power supply and a power amplifier which mount to the main chassis. The power supply is compsoed of two major functional sections; the high-voltage section and the low-voltage section. Each section contains its own power transformer, control interlock, and monitoring circuits. The unit is isolator mounted and is located in an equipment bay.

WRA No. 85 is a power supply which receives 115 volts, 400 Hz., three phase and 28 volts DC from the aft main circuit breaker panel. It is hard mounted in the left wheel well and has three electrical connectors, an elapsed-time meter, and overheat indicator-reset pushbutton. The power supply consists of a blower, a power supply circuit, and an overheat latching relay.

WRA No. 86 is a 5-volt power supply which is the voltage source for various computer subassemblies. It is forced air cooled and is located in the fuselage equipment bay.

Electro Mechanical Devices - This category consists of items such as sensors, accelerometers, etc.

WRA Nos. 87, 88 and 89 are sensors which provide roll, pitch and yaw inputs respectively to a flight control computer for stability augmentation and aircraft attitude. The electronics are mounted on a printed circuit card. Micro electronics and flexible printed cables are used to reduce size and weight.

WRA No. 90 is a digital data recorder which is a 9-track reel-to-reel unit containing 630 feet of 0.5 inch Mylar-base magnetic tape. The unit is forced air cooled and isolator mounted and is located in the fuselage.

WRA No. 91 is a lateral accelerometer assembly which senses acceleration vectors and provides signals to other flight control components. The unit is a conventional force rebalance unit. Pendulum and suspension are fabricated of quartz fibers and a thin film of silver is vapor deposited over the pendulum and suspension. The unit is located in the fuselage.

Enclosures - This group includes items such as racks and cabinets.

WRA No. 92 is a rack which provides mounting and electrical I/O control and primary power connections to the rest of the system and required interfacing equipments. The entire unit consists only of the required mechanical structural parts, connectors and interconnecting cabling and is located in an equipment bay.

WRA No. 93 is a cabinet which houses various components of a computer set. The unit consists of structural parts, connectors, and interconnecting cabling and is located in an equipment bay.

WRA No. 94 is a rack which provides mounting and electrical I/O connections for a communications subsystem. The unit consists only of the required mechanical structural parts, connectors, and interconnecting cabling. It is located in an equipment bay.

WRA No. 95 is a cabinet which houses various components of a computer subsystem. The unit consists of structural parts, connectors, and interconnecting cabling and is located in an equipment bay.

APPENDIX B

Thermal and Pressure Data

The following Tables present the pertinent thermal and environmental data experienced during demonstration tests and field operations for each WRA. The air ambient temperature parameters which include the range, exposure duration, and rate of change for all items are shown. The cooling air characteristics (i.e., temperature, flow rates, rates of change thereof and extent of exposure) for those WRA's that require supplemental cooling are also given. The absolute pressure levels experienced during the laboratory test and in a typical mission and the associated rates of change are summarized.

TABLE B-1 AIR AMBIENT TEMPERATURE PARAMETERS

WBA NO.	RANGE °F LAB	RANGE °F FIELD	RANGE Δ°F MIN	RANGE Δ°F MAX	DURATION-HI-TEMP. HRS. LAB	DURATION-HI-TEMP. HRS. FIELD	DURATION-HI-TEMP. Δ HRS.	DURATION-LO-TEMP. HRS LAB	DURATION-LO-TEMP. HRS FIELD	DURATION-LO-TEMP. Δ HRS.	RATE OF CHANGE °/MIN. LAB	RATE OF CHANGE °/MIN. FIELD	RATE OF CHANGE Δ °/MIN.
1	-65 to +131	+50 to +160	(-)115	+29	0	5.6	+5.6	19.4	0	-19.4	9	10	+1.0
2	-65 to +106	+50 to +170	(-)115	+64	0	10.8	+10.8	36	10.8	-25.2	9.0	10	+1.0
3	-65 to +131	+5 to +105	(-)70	+26	354	56	-298	178	338	+160	9.0	10	+1.0
4	-65 to +131	+50 to +160	(-)115	+29	0	5.6	+5.6	19.4	0	-19.4	9.0	10	+1.0
5	-94 to +91	+12 to +127	(-)106	+35	375	50	-325	58	0	-58	9.0	10	+1.0
6	-65 to +131	+70 to +141	(-)135	+10	51	5.6	-45.4	19.4	0	-19.4	9.0	10	+1.0
7	-65 to +131	+5 to +105	(-)70	+26	354	56	-298	178	338	+160	9.0	10	+1.0
8	-65 to +106	+50 to +170	(-)115	+64	0	10.8	+10.8	36	10.8	-25.2	9.0	10	+1.0
9	+62 to +82	+20 to +126	-42	+44	0	5.5	+5.5	0	5.5	-5.5	0	10	+10
10	-65 to +131	+70 to +141	(-)135	+10	51	5.6	-45.4	19.4	0	-19.4	9.0	10	+1.0
11	-65 to +160	-50 to +180	(-)15	+20	0	66	+66	330	200	-130	9.0	10	+1.0
12	-65 to +160	-50 to +180	(-)15	+20	0	66	+66	330	200	-130	9.0	10	+1.0
13	-65 to +131	+5 to +105	(-)70	+26	354	56	-298	178	338	+160	9.0	10	+1.0
14	+58 to +78	+20 to +126	-38	+48	0	42.5	+42.5	0	42.5	+42.5	0	10	+10
15	-31 to +120	+12 to +127	(-)77	+7	51	5.6	-45.4	19.4	0	-19.4	9	10	+1.0
16	-65 to +131	+50 to +160	(-)115	+29	0	5.6	+5.6	29.4	0	-19.4	9	10	+1.0
17	-65 to +131	+50 to +160	(-)115	+29	0	5.6	+5.6	19.4	0	-19.4	9.0	10	+1.0
18	-31 to +120	+12 to +127	(-)77	+7	0	5.6	+5.6	19.4	0	-19.4	9.0	10	+1.0
19	-65 to +106	+50 to +170	(-)115	+64	0	10.8	+10.8	36	10.8	-25.2	9.0	10	+1.0
20	-94 to +91	+12 to +127	(-)106	+36	375	50	-325	58	0	-58	9.0	10	+1.0
21	+62 to +131	+20 to +126	-42	-5	2.4	5.5	+3.1	4.15	5.5	+1.35	1-2	10	+9 to +8
22	-94 to +91	+12 to +127	(-)106	+36	375	50	-325	58	0	-58	9.0	10	+1.0
23	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
24	-65 to +160	+5 to +105	(-)70	-55	526	105	-421	306	105	-201	9.0	10	+1.0
25	-31 to +120	+60 to +80	(-)91	-40	51	0	-51	19.4	0	-19.4	9.0	10	+1.0
26	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
27	-65 to +160	-60 to +80	(-)125	-80	940	0	-940	158	0	-158	9.0	1.0	-8
28	-65 to +131	+50 to +160	(-)115	+29	0	5.6	+5.6	19.4	0	-19.4	9.0	10	+1.0
29	+58 to +78	+20 to +126	-38	+48	0	5.5	+5.5	0	5.5	+5.5	0	10	+10
30	-65 to +131	+5 to +105	(-)70	-26	354	56	-298	178	338	+160	9.0	10	+1.0
31	-31 to +120	+60 to +80	(-)91	-40	51	0	-51	19.4	0	-19.4	9.0	10	+1.0
32	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
33	+62 to +131	+30 to +126	-42	-5	2.4	5.5	+3.1	4.15	5.5	+1.35	1-2	10	+9 to +8
34	+59 to +77	+20 to +126	-35	+49	0	41	+41	0	41	+41	0	1.0	+1.0
35	-65 to +131	+50 to +160	(-)115	+29	0	175	+175	590	0	-590	9.0	10.0	+1.0
36	+62 to +131	+20 to +126	-42	-5	2.4	5.5	+3.1	4.15	5.5	+1.35	1-2	10	+9 to +8
37	-65 to +160	-60 to +80	(-)125	-80	940	0	-940	158	0	-158	9.0	1.0	-8.0

TABLE B-1 AIR AMBIENT TEMPERATURE PARAMETERS (Continued)

[illegible] NO.	RANGE °F LAB	RANGE °F FIELD	RANGE Δ °F MIN	RANGE Δ °F MAX	DURATION-HI-TEMP. HRS. LAB	DURATION-HI-TEMP. HRS. FIELD	DURATION-HI-TEMP. Δ HRS.	DURATION-LO-TEMP. HRS. LAB	DURATION-LO-TEMP. HRS. FIELD	DURATION-LO-TEMP. Δ HRS.	RATE OF CHANGE °/MIN. LAB	RATE OF CHANGE °/MIN. FIELD	RATE OF CHANGE Δ °/MIN.
38	-65 to +160	+10 to +120	(-)75	-40	224	90	-134	555	270	-285	9.0	10	+1.0
39	-65 to +160	+10 to +120	(-)75	-40	224	90	-134	555	270	-285	9.0	10	+1.0
40	-65 to +160	+10 to +120	(-)75	-40	224	90	-134	555	270	-285	9.0	10	+1.0
41	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
42	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
43	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
44	-94 to +91	-48 to +160	(-)46	+69	0	50	+50	58	25	-33	9	10	+1.0
45	+58 to +78	+20 to +126	-38	+48	0	42.5	+42.5	0	42.5	+42.5	0	10	+10
46	-65 to +160	+10 to +120	(-)75	-40	890	75	-815	324	226	-98	9.0	10	+1
47	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
48	-65 to +160	+10 to +120	(-)75	-40	1450	115	-1335	730	350	-380	9.0	10	+1
49	+58 to +78	+20 to +126	-38	+48	0	42.5	+42.5	0	42.5	+42.5	0	10	+10
50	-65 to +131	+50 to +160	(-)115	+29	0	5.6	+5.6	19.4	0	-19.4	9	10	+1.0
51	+15 to +131	+20 to +126	+10	-5	570	85	-485	680	85	-595	1-4	10	+9 to +6
52	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
53	-65 to +160	+30, to +120	(-)95	-40	224	90	-134	555	270	-285	9.0	1.0	-8.0
54	-31 to +120	+60 to +80	(-)91	-40	51	0	-51	19.4	0	-19.4	9.0	10	+1.0
55	-65 to +160	-60 to +80	(-)125	-80	940	0	-940	158	0	-158	9.0	1.0	-8.0
56	-31 to +120	+60 to +80	(-)91	-40	51	0	-51	19.4	0	-19.4	9.0	10	+1.0
57	-65 to +160	+30 to +120	(-)95	-40	224	90	-134	555	270	-285	9.0	1.0	-8.0
58	-31 to +120	+60 to +80	(-)91	-40	51	0	-51	19.4	0	-19.4	9.0	10	+1.0
59	-65 to +160	+60 to +80	(-)125	-80	580	0	-580	88	0	-88	9.6	1.0	-8.6
60	-31 to +120	+60 to +80	(-)91	-40	51	0	-51	19.4	0	-19.4	9	10	+1.0
61	-65 to +160	+30 to +120	(-)	-40	1450	115	-1335	730	350	-380	9.0	1.0	-8.0
62	-31 to +120	+60 to +80	(-)91	-40	51	0	-51	19.4	0	-19.4	9.0	10	+1.0
63	+62 to +131	+20 to +126	-42	-5	2.4	5.5	+3.1	4.15	5.5	+1.35	1-2	10	+9 to +8
64	-31 to +120	+60 to +80	(-)91	-40	51	0	-51	19.4	0	-19.4	9	10	+1.0
65	-65 to +160	+30 to +120	(-)95	-40	1450	115	-1335	730	350	-380	9.0	1.0	-8.0
66	-94 to +91	-60 to +80	(-)154	-11	375	0	-375	58	0	-58	9.0	10	+1.0
67	-31 to +120	-60 to +80	(-)91	-40	51	0	-51	19.4	0	-19.4	9.0	10	+1.0
68	-65 to +131	+60 to +80	(-)125	-51	354	0	-354	178	0	-178	9.0	1.0	-8.0
69	+62 to +131	+20 to +126	-42	-5	2.4	5.5	+3.1	4.15	5.5	+1.35	1-2	10	+9 to +8
70	-31 to +120	+50 to +160	(-)81	+40	51	0	-51	19.4	0	-19.4	9	10	+1.0
71	-65 to +160	+30 to +120	(-)95	-40	600	66	-534	450	200	-250	9.0	1.0	-8.0
72	-31 to +120	+60 to +80	(-)91	-40	51	0	-51	19.4	0	-19.4	9.0	10	+1.0
73	-65 to +131	+60 to +80	(-)125	-51	354	0	-354	178	0	-178	9.0	1.0	-8.0
75	-31 to +120	+60 to +80	(-)91	-40	51	0	-51	19.4	0	-19.4	9.0	10	+1.0

TABLE B-1 AIR AMBIENT TEMPERATURE PARAMETERS (Continued)

WRA NO.	RANGE °F LAB	RANGE °F FIELD	RANGE Δ °F MIN	RANGE Δ °F MAX	DURATION-HI-TEMP. HRS. LAB	DURATION-HI-TEMP. HRS. FIELD	DURATION-HI-TEMP. Δ HRS.	DURATION-LO-TEMP. HRS. LAB	DURATION-LO-TEMP. HRS. FIELD	DURATION-LO-TEMP. Δ HRS.	RATE OF CHANGE °/MIN. LAB	RATE OF CHANGE °/MIN. FIELD	RATE OF CHANGE Δ °/MIN.
75	-65 to +131	+60 to +80	(-)125	-51	354	0	-354	178	0	-178	9.0	1.0	-8.0
76	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
77	-31 to +120	+60 to +80	(-)91	-40	51	0	-51	19.4	0	-19.4	9.0	10	+1.0
78	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
79	-65 to +131	+50 to +160	(-)115	+29	0	5.6	+5.6	19.4	0	-19.4	9	10	+1.0
80	-65 to +160	+10 to +120	(-)75	-40	0	66	+66	330 ≤ 50	200 ≤ 50	-130	9.0	10	+1.0
81	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
82	-65 to +131	+5 to +105	(-)70	-26	354	56	-298	178	338	+160	9.0	10	+1.0
83	-65 to +131	+12 to +127	(-)77	-4	51	5.6	-45.4	19.4	0	-19.4	9.0	10	+1.0
84	+62 to +82	+20 to +126	-42	+44	0	5.5	+5.5	0	5.5	+5.5	0	10	+10
85	-94 to +91	+12 to +127	(-)106	+35	375	25	-350	58	0	-58	9	10	+1.0
86	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
87	-65 to +160	+10 to +120	(-)75	-40	224	90	-134	555	270	-285	9.0	10	+1.0
88	-65 to +160	+10 to +120	(-)75	-40	224	90	-134	555	270	-285	9.0	10	+1.0
89	-65 to +160	+10 to +120	(-)75	-40	224	90	-134	555	270	-285	9.0	10	+1.0
90	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0
91	-65 to +160	+10 to +120	(-)75	-40	224	90	-134	555	270	-285	9.0	10	+1.0
92	+62 to +131	+20 to +125	-42	-5	2.4	5.5	+3.1	4.15	5.5	+1.35	1-2	10	+9 to +8
93	+59 to +77	+20 to +126	-39	+40	0	41	+41	0	41	+41	0	1.0	+1.0
94	+62 to +131	+20 to +126	-42	-5	2.4	5.5	+3.1	4.15	5.5	+1.35	1-2	10	+9 to +8
95	+59 to +77	+20 to +126	-39	+49	0	41	+41	0	41	+41	0	1.0	+1.0

TABLE 8-2 EQUIPMENT COOLING AIR PARAMETERS

MRA NO.	TEMPERATURE RANGE °F LAB	TEMPERATURE RANGE °F FIELD	Δ - °F MIN	Δ - °F MAX	TEMP. - RATE OF CHANGE °/MIN LAB	TEMP. - RATE OF CHANGE °/MIN FIELD	TEMP. - RATE OF CHANGE Δ °/MIN	FLOW - RATE OF CHANGE #/MIN/MIN LAB	FLOW - RATE OF CHANGE #/MIN/MIN FIELD	FLOW - RATE OF CHANGE Δ #/MIN/MIN	FLOW - RANGE #/MIN LAB	FLOW - RANGE #/MIN FIELD	FLOW - RANGE Δ - #/MIN MIN	FLOW - RANGE Δ - #/MIN MAX	DURATION HOURS LAB	DURATION HOURS FIELD	DURATION Δ HRS
1	+70 to +80	+70 to +80	0	0	0	0	0	0	0	0	0.60	0.61	-	+0.01	85	113	+28
2	+93 CONST	+3 to +166	-90	+73	0	0	0	0	0	0	5	3.27 to 20.25	-1.73	+15.25	162	21	-141
4	+70 to +80	+70 to +80	0	0	0	0	0	0	0	0	0.60	0.66	-	+0.06	85	113	+28
6	0 to +80	+70 to +80	+70	0	0	0	0	0	0	0	0.55	0.55	-	0	85	113	+28
7	+75 to +85	+60 to +80	-15	-5	INSTAN.	1.0	> -1	0.60	0.60	0	1.8 to 2.2	1.1 to 2.2	-0.7	0	382	85	-297
8	+93 CONST	+3 to +166	-90	+73	0	0	0	0	0	0	5	3.27 to 20.25	-1.73	+15.25	162	21	-141
9	+35 to +115	+60 to +70	+25	-45	NEGLIG.	NEGLIG.	0	NEGLIG.	NEGLIG.	0	0.68	0.68	-	0	111	111	0
10	+70 to +80	+70 to +80	0	0	0	0	0	0	0	0	0.55	0.55	-	0	85	113	+28
13	+75 to +85	+60 to +80	-15	-5	INSTAN.	1.0	> -1	0	0.05	+0.05	0.28 to 0.34	0.16 to 0.30	-0.12	-0.04	382	85	-297
14	+70 to +115	+60 to +70	-10	-45	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	1.16 to 1.4	0.52	-0.64	-0.88	850	850	0
17	+70 to +80	+70 to +80	0	0	0	0	0	0	0	0	0.36	0.59	-	+0.23	85	113	+28
19	+93 CONST.	+3 to +166	-90	+73	0	0	0	0	0	0	5	3.27 to 20.25	-1.73	+15.25	162	21	-141
23	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
24	+55 to +120	+40 to +60	+5	-60	INSTAN.	10	> -10	0.26	0.26	0	0.4 to 1.2	1.10 to 1.45	+0.70	+0.25	485	63	-422
26	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
27	-65 to +120	+40 to +60	(-)105	-60	2.0	10	+8	0.12	0.12	0	0.3 to 1.2	0.53 to 0.72	+0.23	-0.48	880	285	-595
28	+70 to +80	+70 to +80	0	0	0	0	0	0	0	0	0.89	0.89	-	0	85	113	+28
29	+35 to +115	+60 to +70	+25	-45	NEGLIG.	NEGLIG.	0	NEGLIG.	NEGLIG.	0	6.4	6.4	-	0	111	111	0
30	+75 to +85	+60 to +80	-15	-5	INSTAN.	1.0	> -1	0.10	0.10	0	0.135 to 0.165	0.13 to 0.32	-0.005	+0.155	382	163	-219
32	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
34	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
37	-65 to +120	+40 to +60	(-)105	-60	2.0	10	+8	0.30	0.30	0	0.6 to 2.4	1.35 to 1.85	+0.75	-0.55	880	285	-595
41	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
42	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
43	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
45	+70 to +115	+60 to +70	-10	-45	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	1.16 to 1.4	0.52	-0.64	-0.88	850	850	0
46	+80	+59 to +65	-21	-15	NEGLIG.	NEGLIG.	0	NEGLIG.	NEGLIG.	0	0.65	0.52 to 0.598	-.13	-0.06	1130	1512	+382
47	+11 to +70	+50 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0

TABLE B-2 EQUIPMENT COOLING AIR PARAMETERS (Continued)

WRA NO.	TEMPERATURE RANGE °F LAB	°F FIELD	Δ - °F MIN	Δ - °F MAX	TEMP. - RATE OF CHANGE °/MIN LAB	°/MIN FIELD	Δ °/MIN	FLOW - RATE OF CHANGE #/MIN/MIN LAB	#/MIN/MIN FIELD	Δ #/MIN/MIN	FLOW - RANGE #/MIN LAB	#/MIN FIELD	Δ - #/MIN MIN	Δ - #/MIN MAX	DURATION HOURS LAB	HOURS FIELD	Δ HRS
48	+80	+59 to +65	-21	-15	NEGLIG.	NEGLIG.	0	NEGLIG.	NEGLIG.	0	0.61	0.51 to 0.596	-.10	-0.03	1620	2330	+710
49	+70 to +115	+60 to +70	-10	-45	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	1.16 to 1.4	0.61	-0.55	-0.79	850	850	0
50	+70 to +80	+70 to +80	0	0	0	0	0	0	0	0	1.20	1.20	-	0	85	113	+28
52	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
76	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
78	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
79	+70 to +80	+70 to +80	0	0	0	0	0	0	0	0	1.50	2.80	-	+1.30	85	113	+28
81	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
82	+75 to +85	+60 to +80	-15	-5	INSTAN.	1.0	>-1	0.25	0.25	0	0.81 to 0.99	0.46 to 0.89	-0.35	-0.10	382	85	-297
83	+70 to +80	+70 to +80	0	0	0	0	0	0	0	0	1.00	1.00	-	0	85	113	+28
84	+35 to +115	+60 to +70	+25	-45	NEGLIG.	NEGLIG.	0	NEGLIG.	NEGLIG.	0	3.76	3.76	-	0	111	111	0
86	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
90	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG	0	13 to 14	19.78	+6.78	+5.78	831	831	0
93	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0
95	+11 to +70	+60 to +70	+49	0	0.72 to 7.5	0	-0.72 to -7.5	NEGLIG.	NEGLIG.	0	13 to 14	19.78	+6.78	+5.78	831	831	0

TABLE B-3 ATMOSPHERIC PRESSURE PARAMETERS

WRA NO.	LEVEL			RATE OF CHANGE		
	PSIA		Δ	PSI/MINUTE		Δ
	LAB	FIELD	PSIA	LAB	FIELD	PSI MIN
1	14.7	2.14	-12.56	0	4	4
2	14.7	2.14	-12.56	0	4	4
3	14.7	2.7	-12.0	0	4	4
4	14.7	2.14	-12.56	0	4	4
5	14.7	2.14	-12.56	0	4	4
6	14.7	2.14	-12.56	0	4	4
7	14.7	2.7	-12.0	0	4	4
8	14.7	2.14	-12.56	0	4	4
9	14.7	8.5	-6.2	0	4	4
10	14.7	2.14	-12.56	0	4	4
11	14.7	0.65	-14.05	0	3	3
12	14.7	0.65	-15.05	0	3	3
13	14.7	2.7	-12.0	0	4	4
14	14.7	8.5	-6.2	0	4	4
15	14.7	2.14	-12.56	0	4	4
16	14.7	2.14	-12.56	0	4	4
17	14.7	2.14	-12.56	0	4	4
18	14.7	2.14	-12.56	0	4	4
19	14.7	2.14	-12.56	0	4	4
20	14.7	2.14	-12.56	0	4	4
21	14.7	8.5	-6.2	0	4	4
22	14.7	2.14	-12.56	0	4	4
23	14.7	8.5	-6.2	0	4	4
24	14.7	2.7	-12.0	0	4	4
25	14.7	2.14	-12.56	0	4	4
26	14.7	8.5	-6.2	0	4	4
27	14.7	7.7	-7.0	0	4	4
28	14.7	2.14	-12.56	0	4	4
29	14.7	8.5	-6.2	0	4	4
30	14.7	2.7	-12.0	0	4	4
31	14.7	2.14	-12.56	0	4	4
32	14.7	8.5	-6.2	0	4	4
33	14.7	8.5	-6.2	0	4	4

TABLE B-3 ATMOSPHERIC PRESSURE PARAMETERS (Continued)

WRA NO.	LEVEL			RATE OF CHANGE		
	PSIA		Δ	PSI/MINUTE		Δ
	LAB	FIELD	PSIA	LAB	FIELD	PSI/MIN
34	14.7	8.5	-6.2	0	4	4
35	14.7	2.14	-12.56	0	4	4
36	14.7	8.5	-6.2	0	4	4
37	14.7	7.7	-7.0	0	4	4
38	14.7	0.65	-14.05	0	3	3
39	14.7	0.65	-14.05	0	3	3
40	14.7	0.65	-14.05	0	3	3
41	14.7	8.5	-6.2	0	4	4
42	14.7	8.5	-6.2	0	4	4
43	14.7	8.5	-6.2	0	4	4
44	14.7	2.14	-12.56	0	4	4
45	14.7	8.5	-6.2	0	4	4
46	14.7	0.65	-14.05	0	3	3
47	14.7	8.5	-6.2	0	4	4
48	14.7	0.65	-14.05	0	3	3
49	14.7	8.5	-6.2	0	4	4
50	14.7	2.14	-12.56	0	4	4
51	14.7	8.5	-6.2	0	4	4
52	14.7	8.5	-6.2	0	4	4
53	14.7	5.65	-9.05	0	3	3
54	14.7	2.14	-12.56	0	4	4
55	14.7	7.7	-7.0	0	4	4
56	14.7	2.14	-12.56	0	4	4
57	14.7	5.65	-9.05	0	3	3
58	14.7	2.14	-12.56	0	4	4
59	14.7	7.7	-7.0	0	4	4
60	14.7	2.14	-12.56	0	4	4
61	14.7	5.65	-9.05	0	3	3
62	14.7	2.14	-12.56	0	4	4
63	14.7	8.5	-6.2	0	4	4
64	14.7	2.14	-12.56	0	4	4
65	14.7	5.65	-9.05	0	3	3
66	14.7	2.14	-12.56	0	4	4

TABLE B-3 ATMOSPHERIC PRESSURE PARAMETERS (Continued)

WRA NO.	LEVEL			RATE OF CHANGE		
	PSIA		Δ	PSI/MINUTE		Δ
	LAB	FIELD	PSIA	LAB	FIELD	PSI/MIN
67	14.7	2.14	-12.56	0	4	4
68	14.7	7.7	-7.0	0	4	4
69	14.7	8.5	-6.2	0	4	4
70	14.7	2.14	-12.56	0	4	4
71	14.7	5.65	-9.05	0	4	4
72	14.7	2.14	-12.56	0	3	3
73	14.7	7.7	-7.0	0	4	4
74	14.7	2.14	-12.56	0	4	4
75	14.7	7.7	-7.0	0	4	4
76	14.7	8.5	-6.2	0	4	4
77	14.7	2.14	-12.56	0	4	4
78	14.7	8.5	-6.2	0	4	4
79	14.7	2.14	-12.56	0	4	4
80	14.7	0.65	-14.05	0	4	4
81	14.7	8.5	-6.2	0	3	3
82	14.7	2.7	-12.0	0	4	4
83	14.7	2.14	-12.56	0	4	4
84	14.7	8.5	-6.2	0	4	4
85	14.7	2.14	-12.56	0	4	4
86	14.7	8.5	-6.2	0	4	4
87	14.7	0.65	-14.05	0	4	4
88	14.7	0.65	-14.05	0	3	3
89	14.7	0.65	-14.05	0	3	3
90	14.7	8.5	-6.2	0	3	3
91	14.7	0.65	-14.05	0	4	4
92	14.7	-	-	0	3	3
93	14.7	8.5	-6.2	-	-	-
94	14.7	-	-	0	4	4
95	14.7	8.5	-6.2	-	-	-
				0	4	4

APPENDIX C

Vibration Data

The following Figures and Tables present the demonstration and field vibration data and the results of analysis of actual flight data. Power Spectral Density (PSD) plots for various flight stations in the three gas turbine jet aircraft for different flight conditions are shown. Acceleration versus frequency plots for the turboprop (prop jet) aircraft for two flight conditions and three flight stations are given. A summary of the type and duration of each WRA's vibration environment during its test and mission is presented.

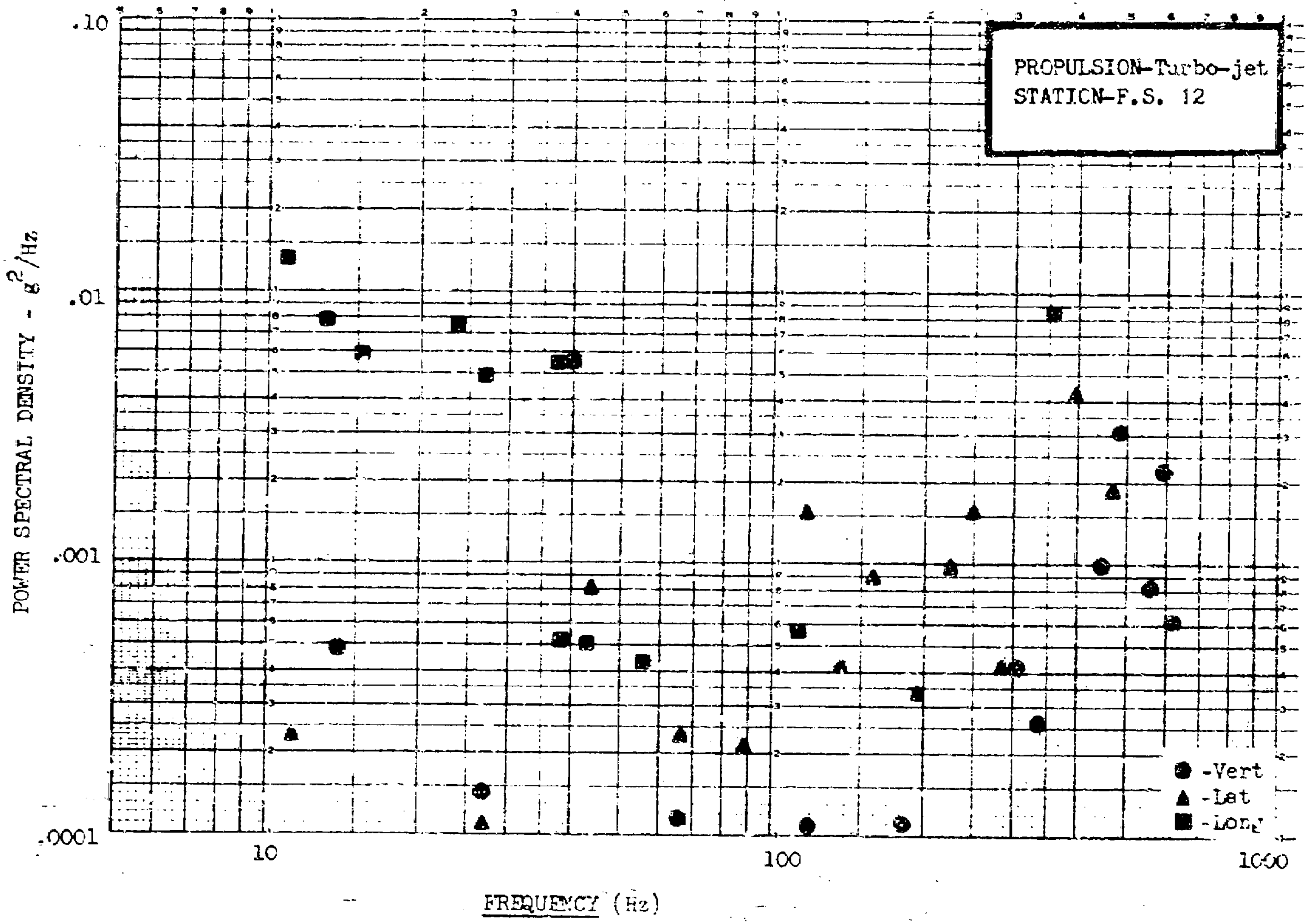

FIGURE C-1 PSD PLOT FOR AIRCRAFT NO. 1

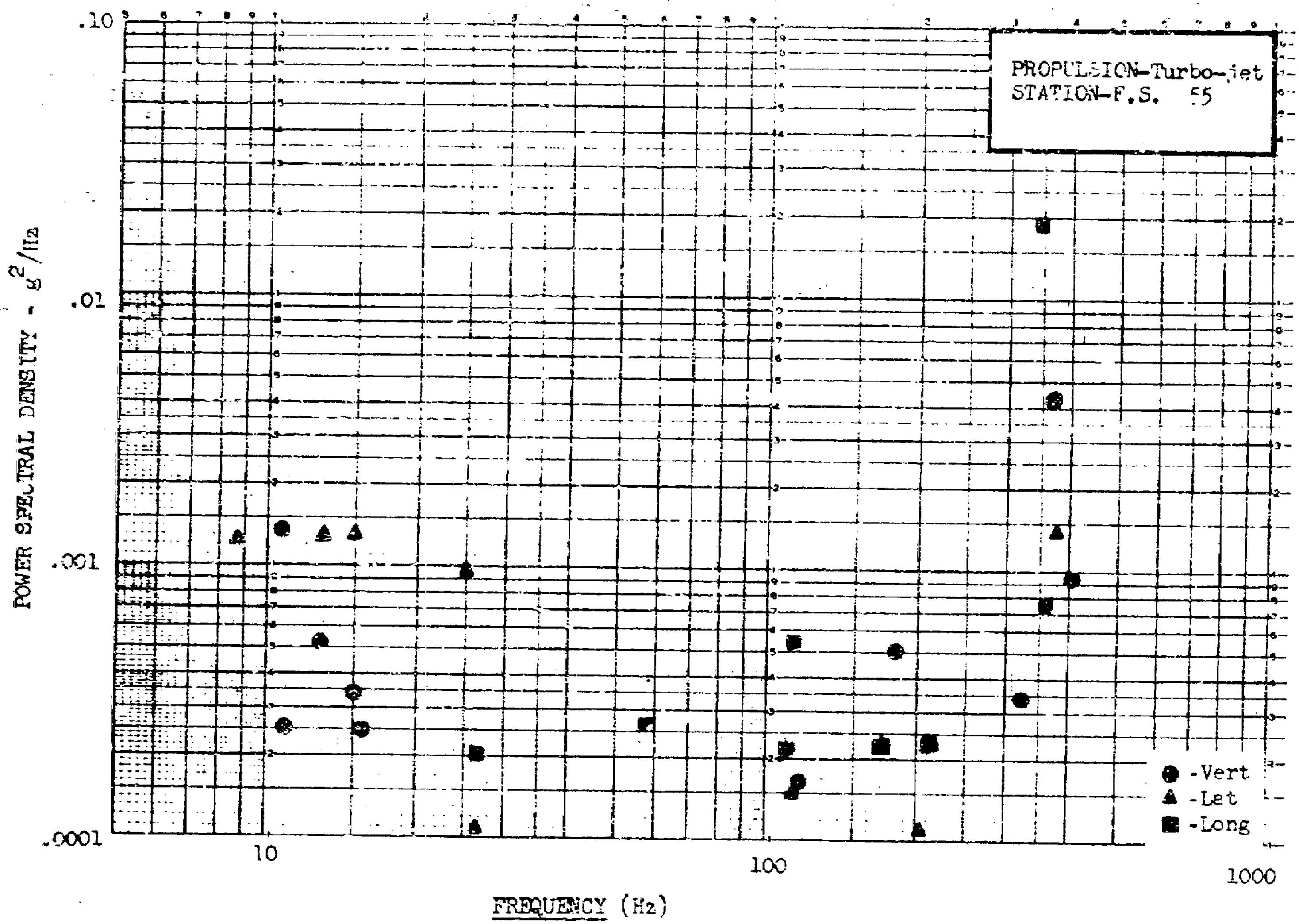

FIGURE C-2 PSD PLOT FOR AIRCRAFT NO. 1

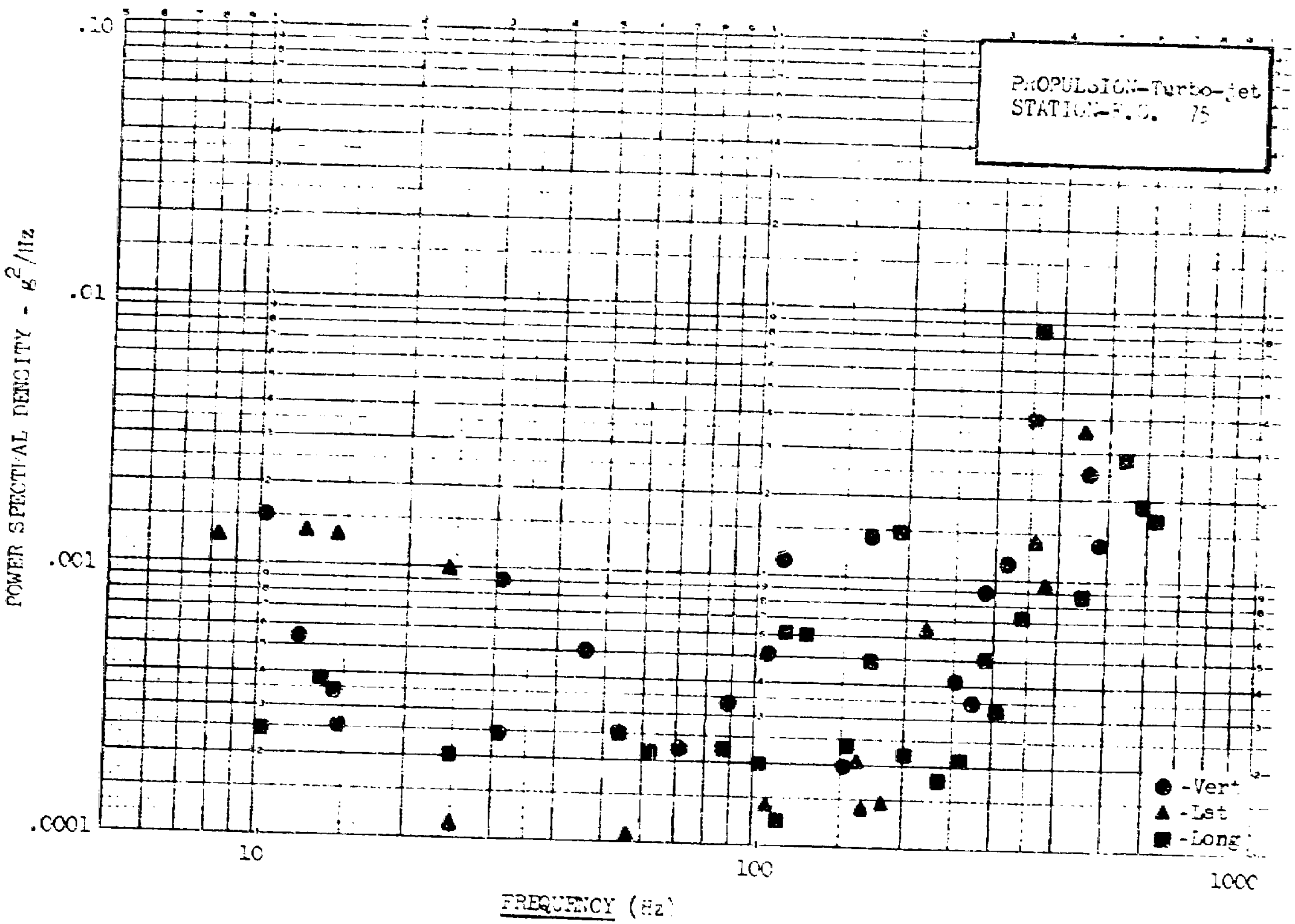

FIGURE C-3 PSD PLOT FOR AIRCRAFT NO. 1

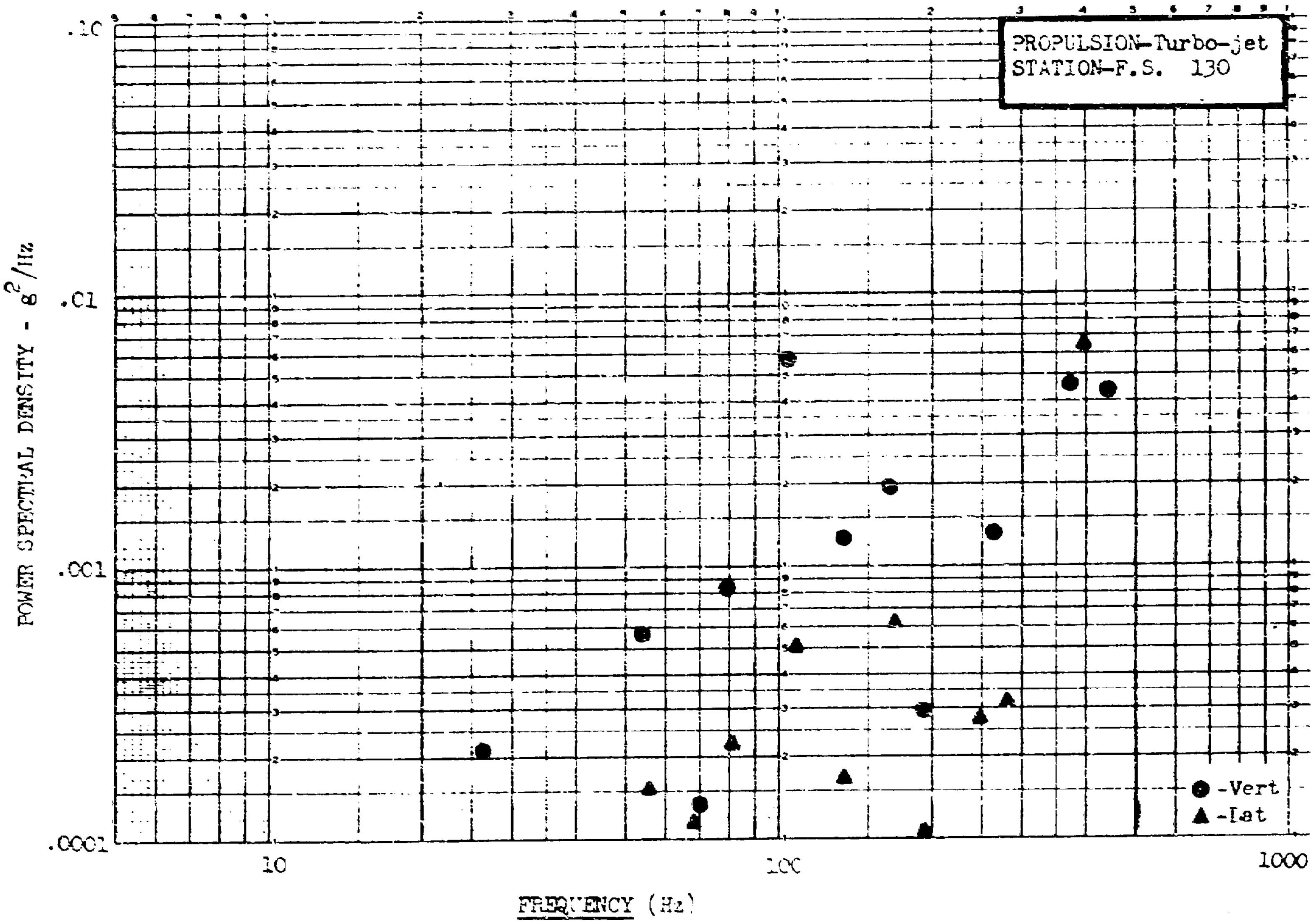

FIGURE C-4 PSD PLOT FOR AIRCRAFT NO. 1

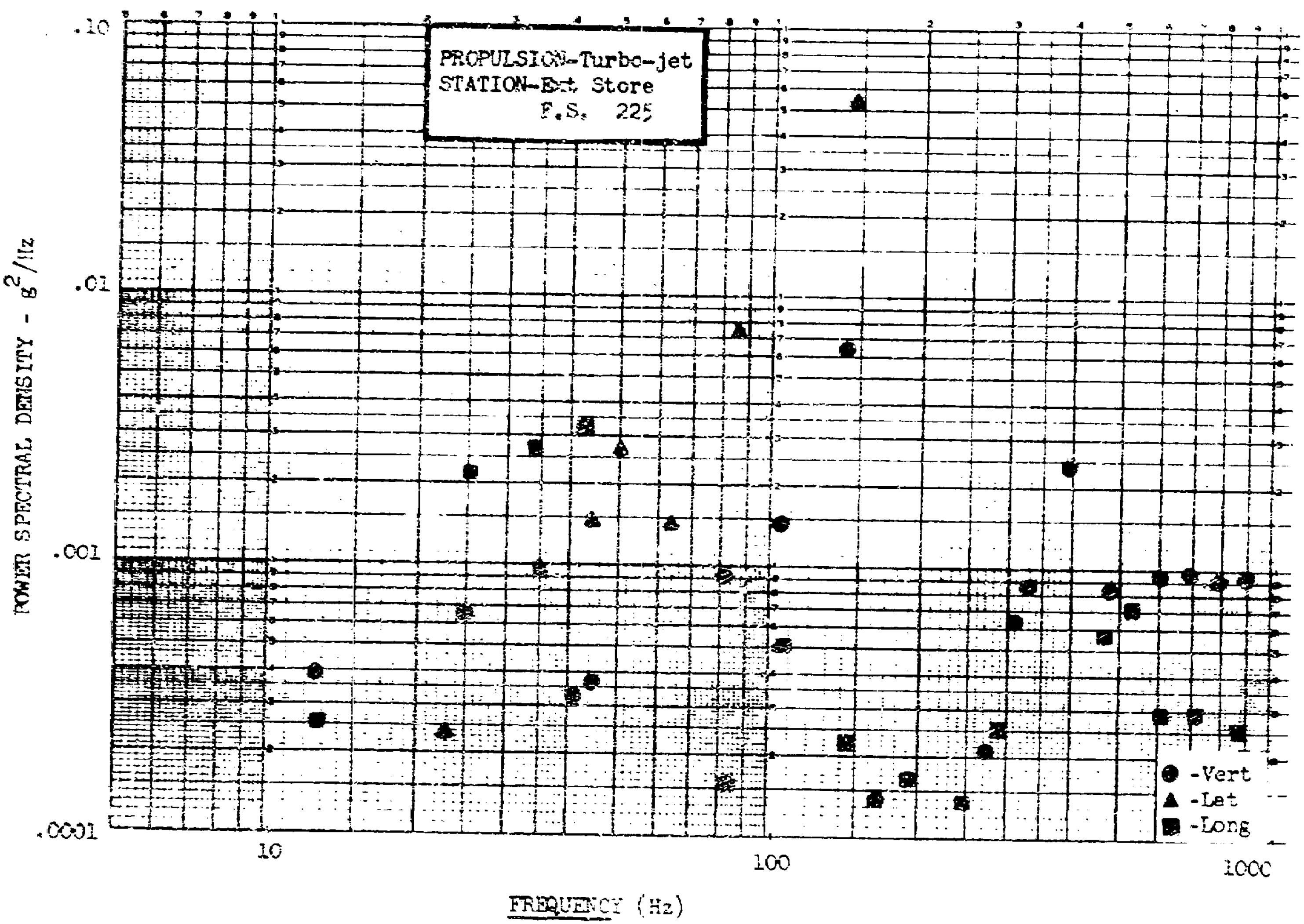

FIGURE C-5 PSD PLOT FOR AIRCRAFT NO. 1

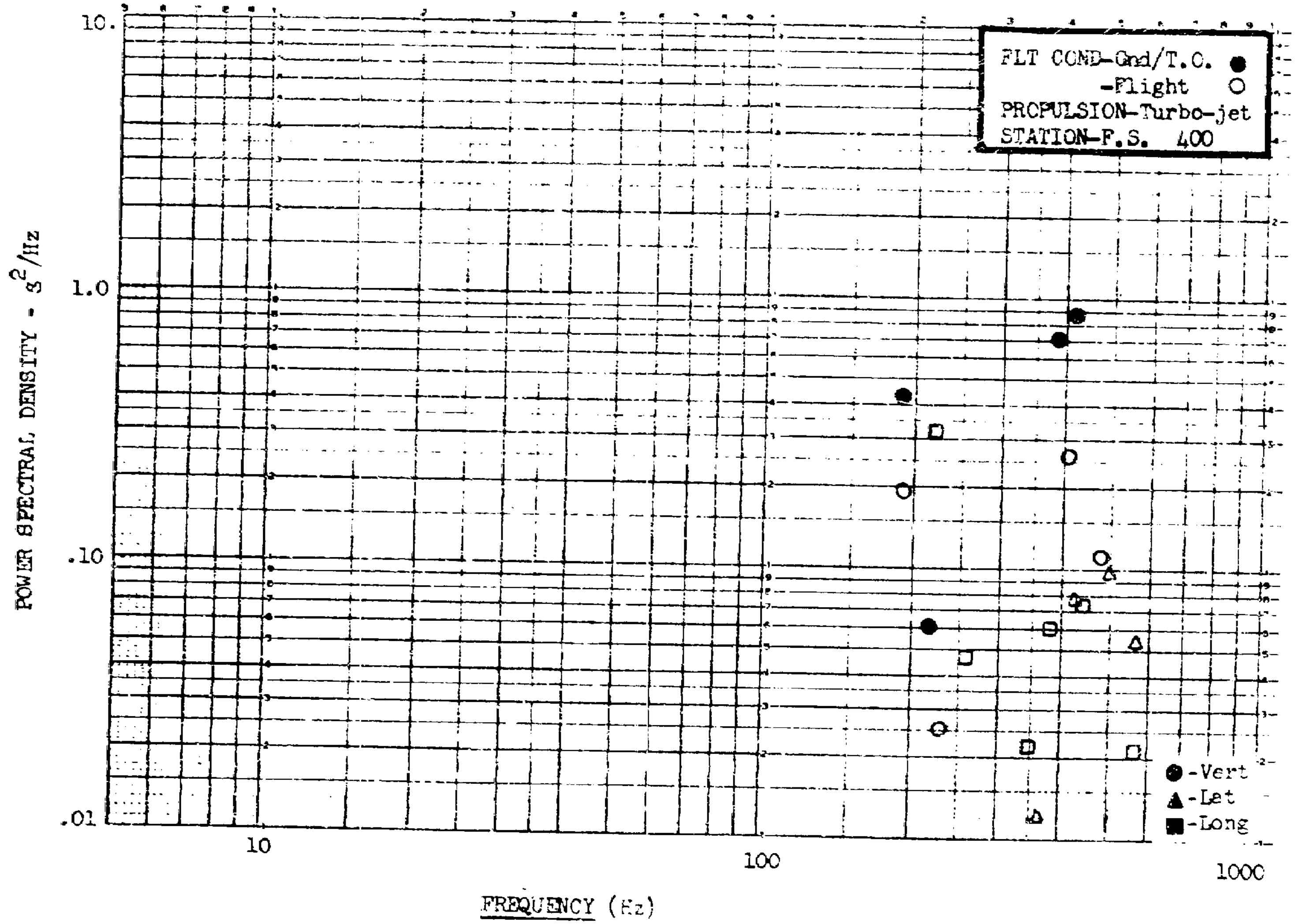

FIGURE C-6 PSD PLOT FOR AIRCRAFT NO. 1

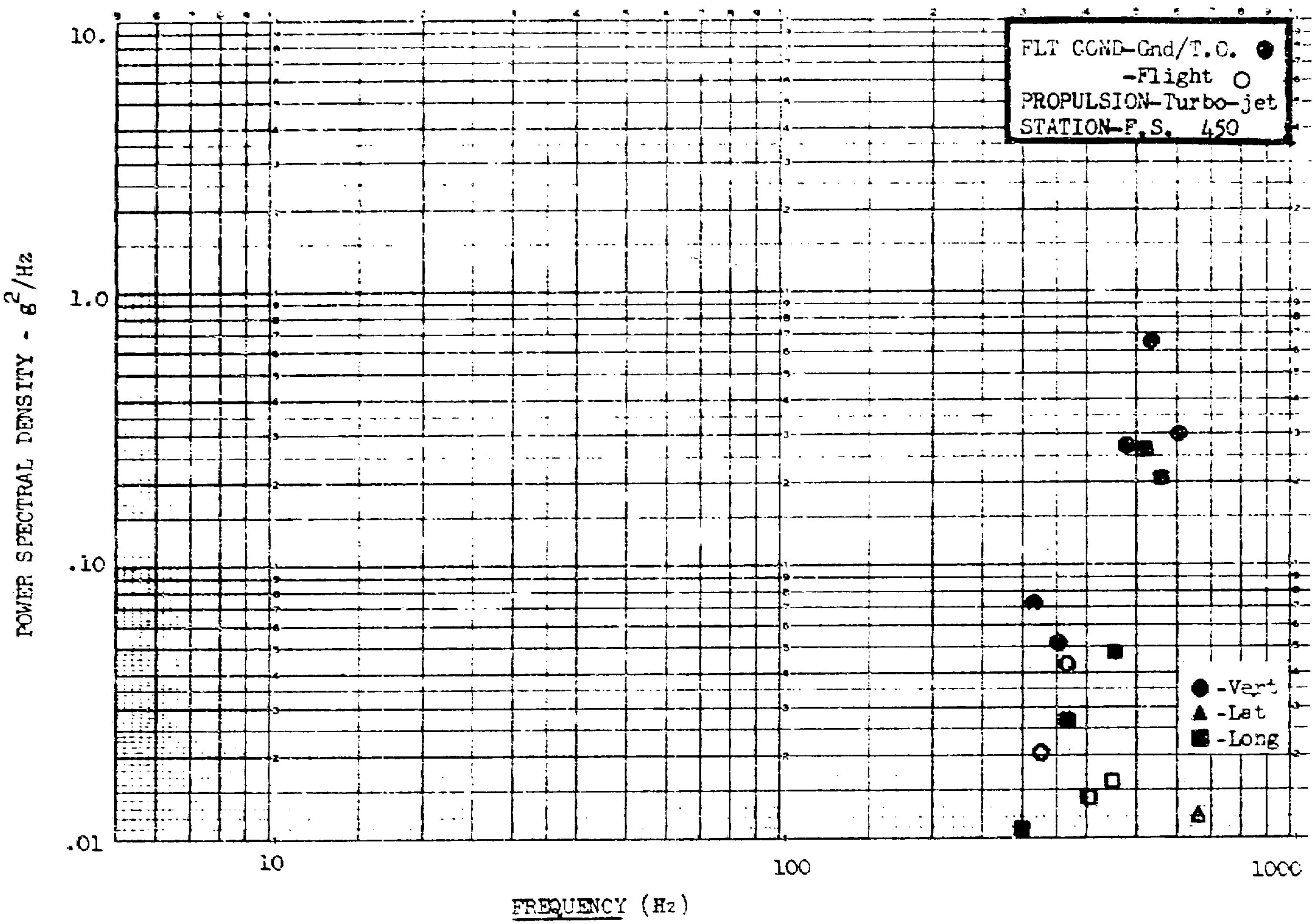

FIGURE C-7 PSD PLOT FOR AIRCRAFT NO. 1

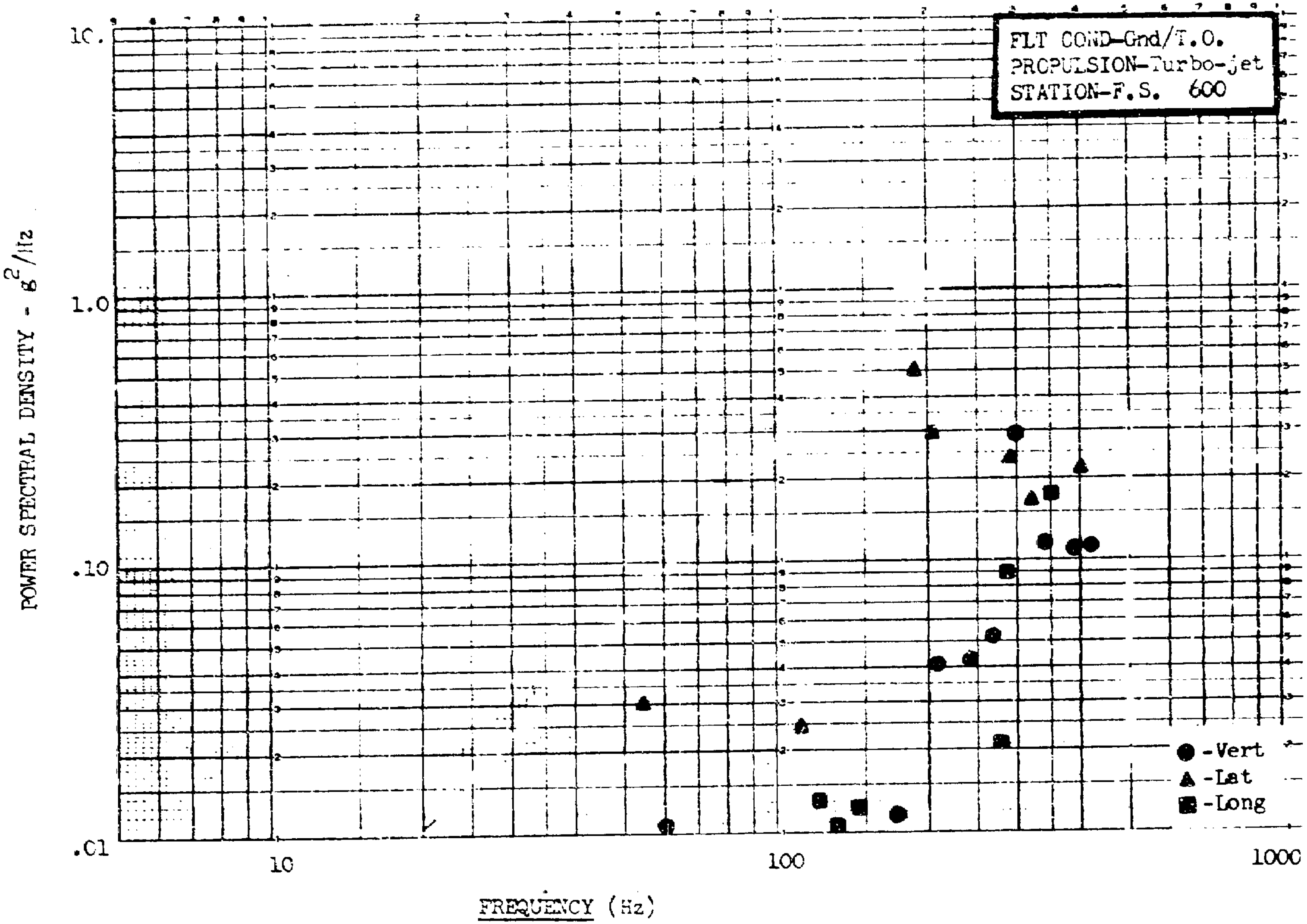

FIGURE C-8 PSD PLOT FOR AIRCRAFT NO. 1

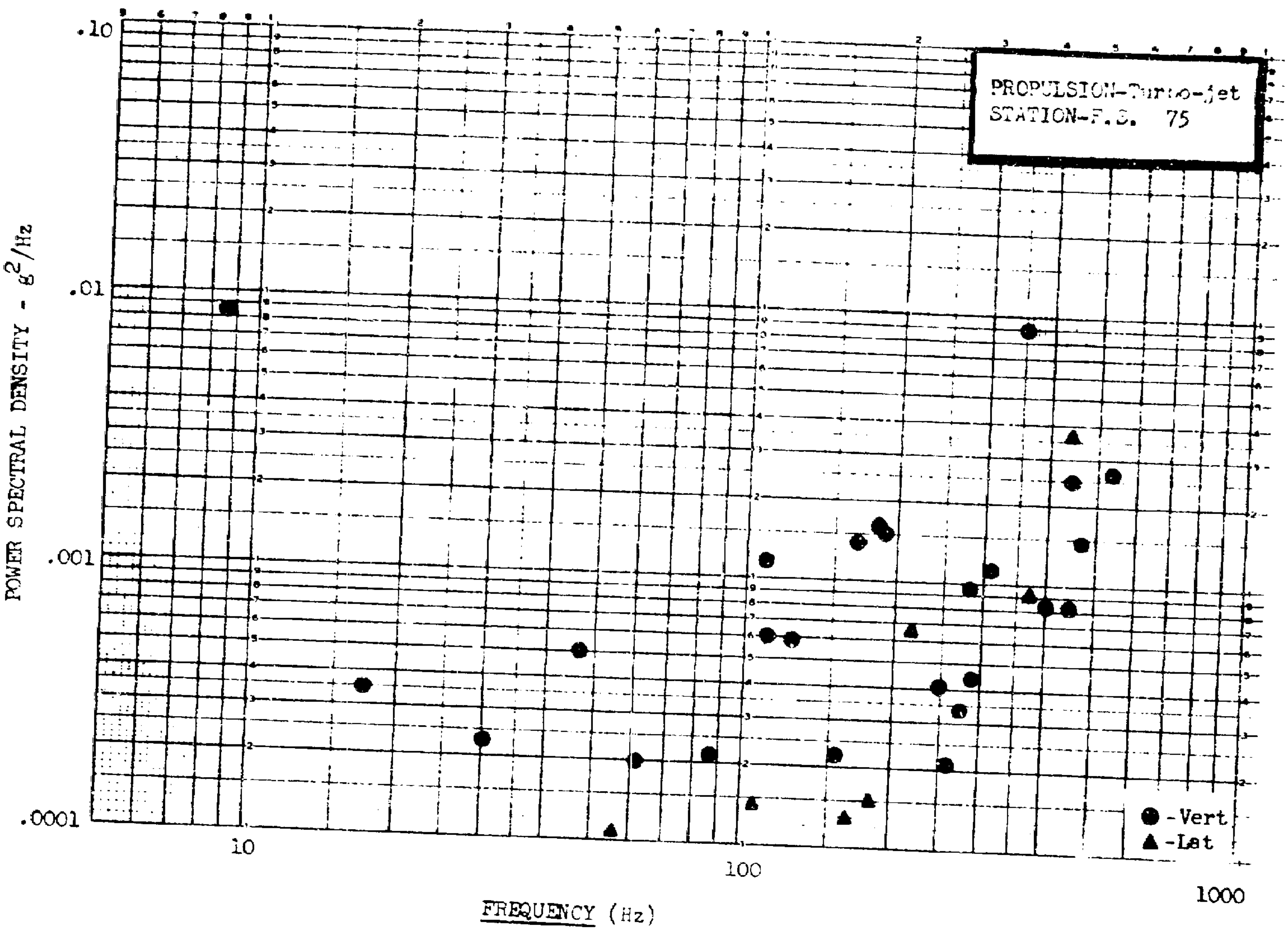

FIGURE C-9 PSD PLOT FOR AIRCRAFT NO. 2

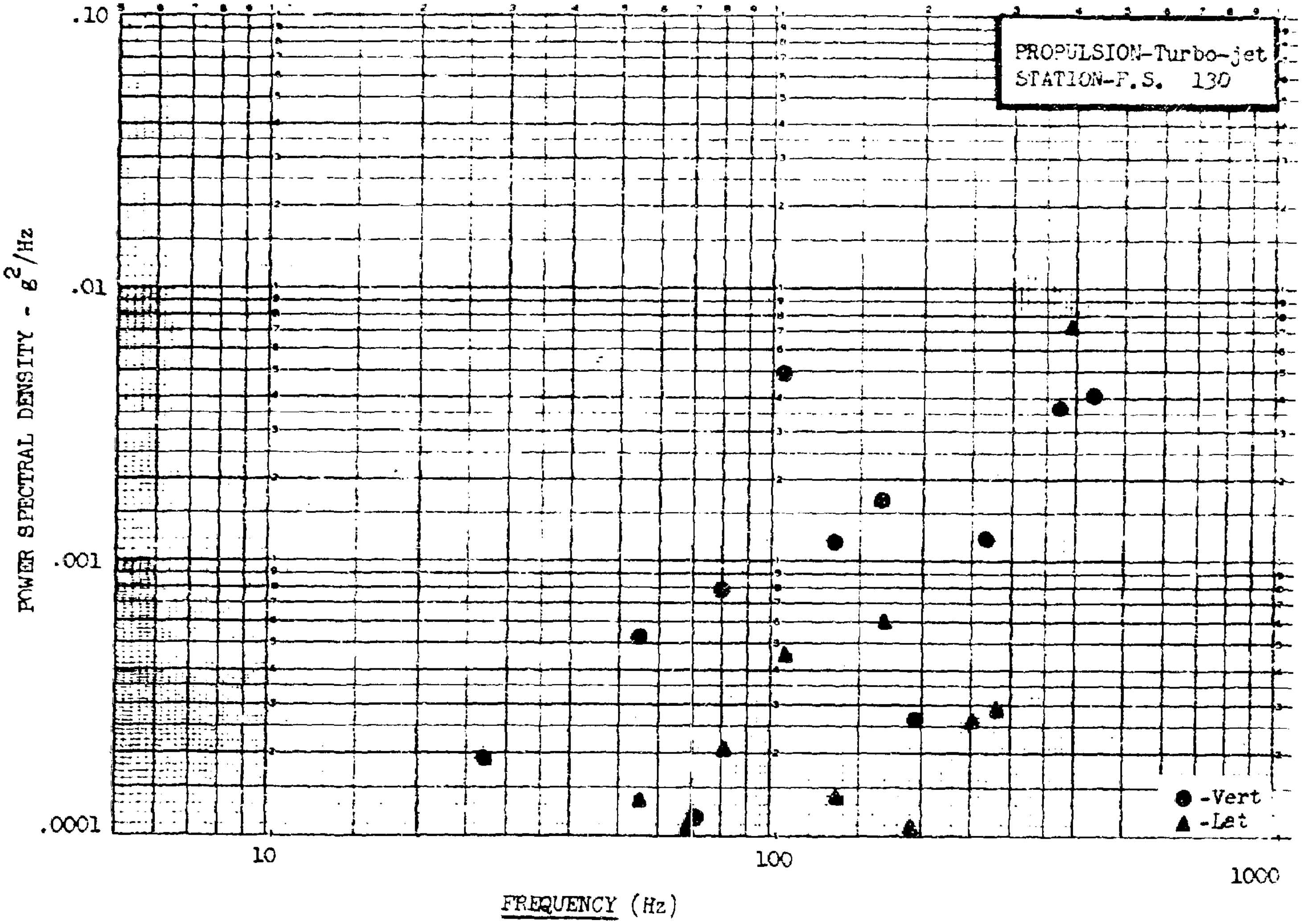

FIGURE C-10 PSD PLOT FOR AIRCRAFT NO. 2

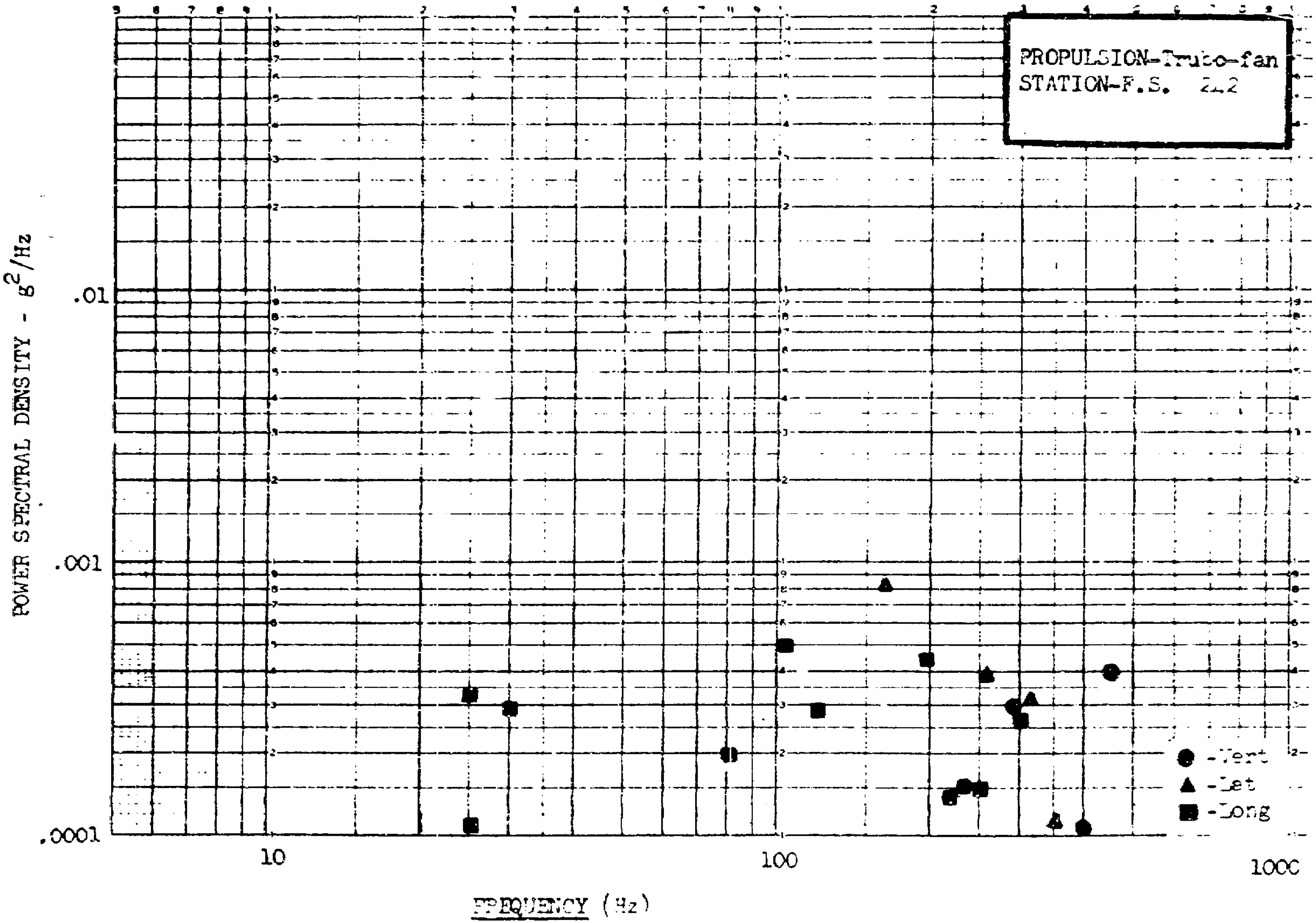

FIGURE C-11 PSD PLOT FOR AIRCRAFT NO. 3

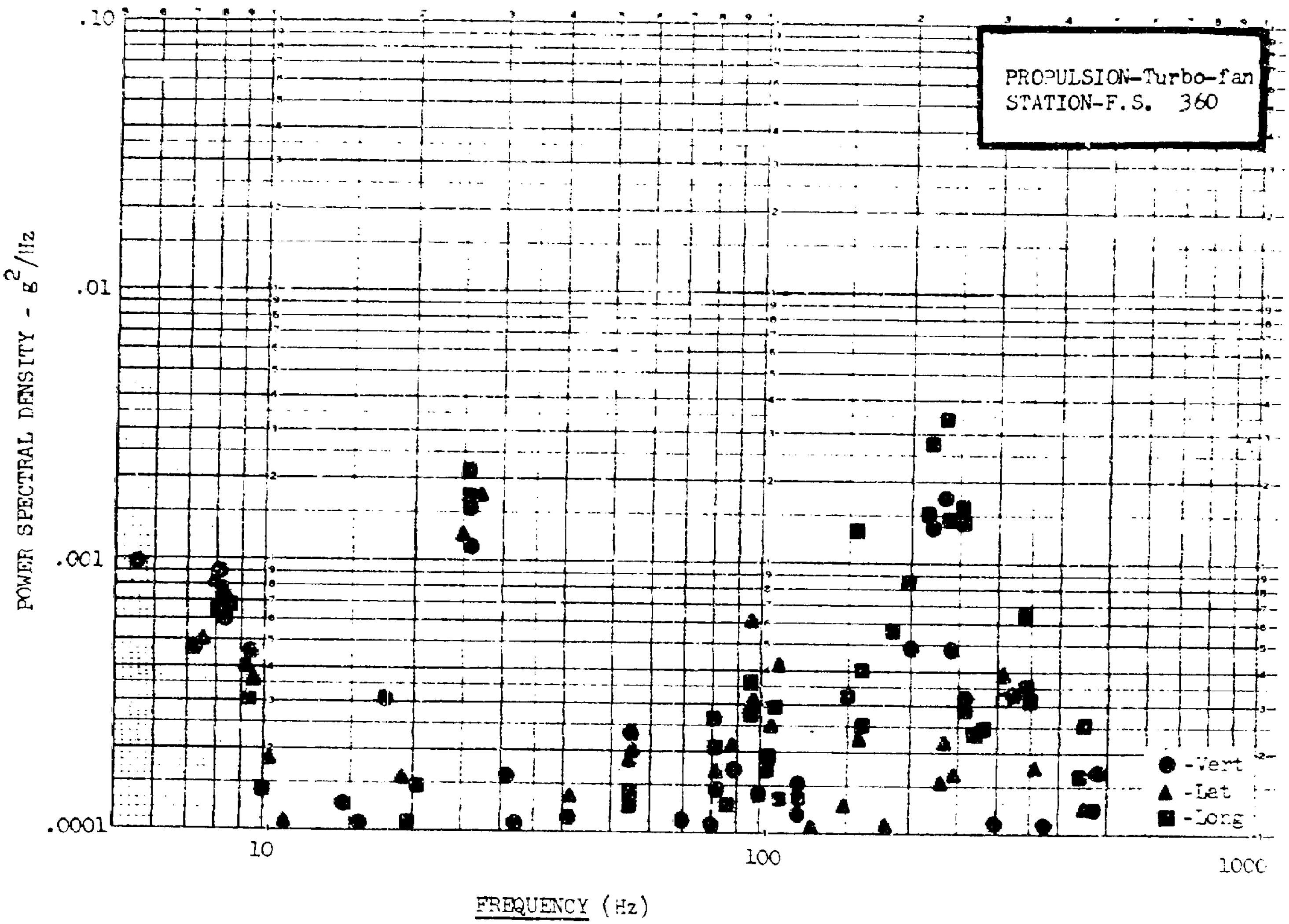

FIGURE C-12 PSD PLOT FOR AIRCRAFT NO. 3

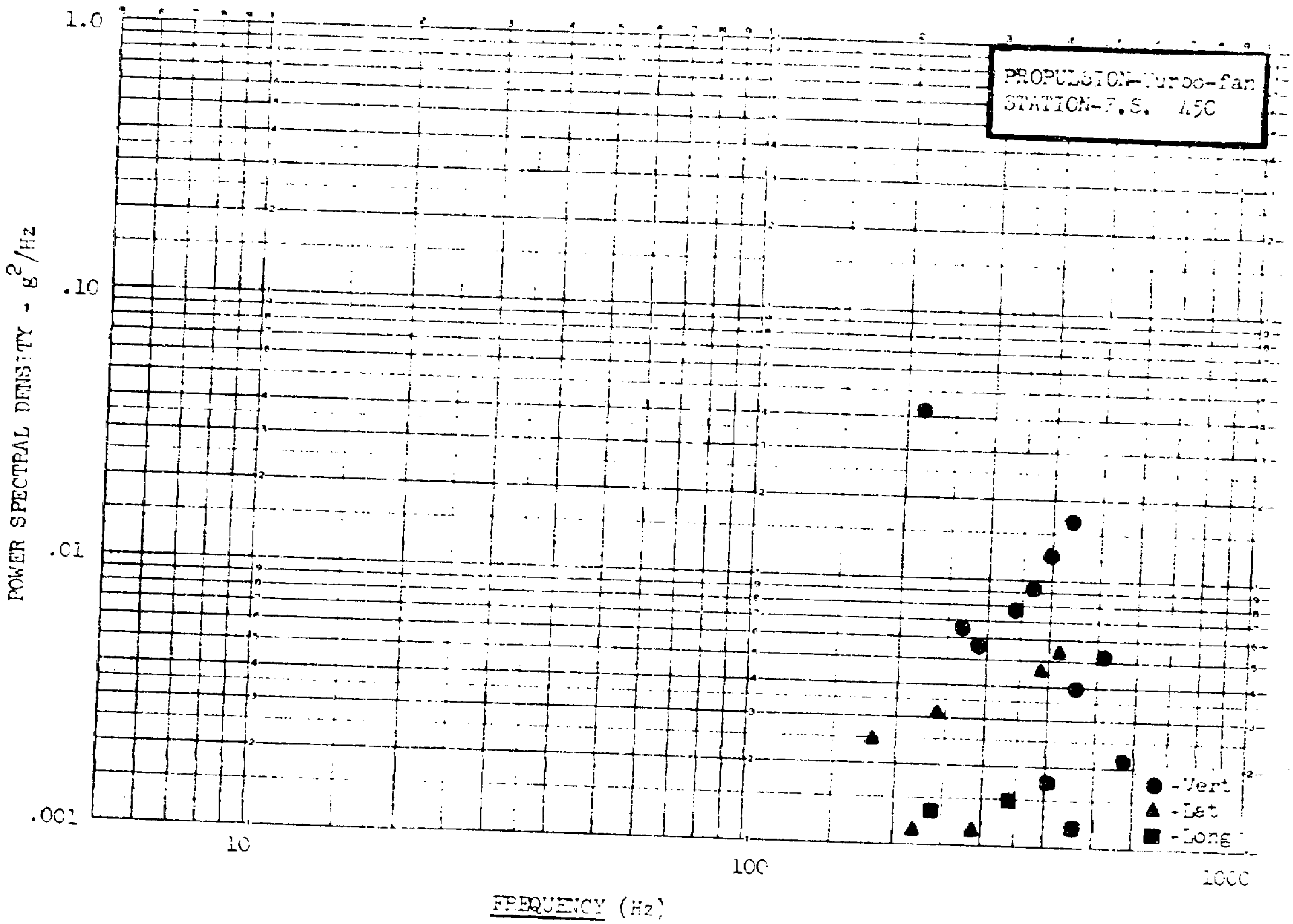

FIGURE C-13 PSD PLOT FOR AIRCRAFT NO. 3

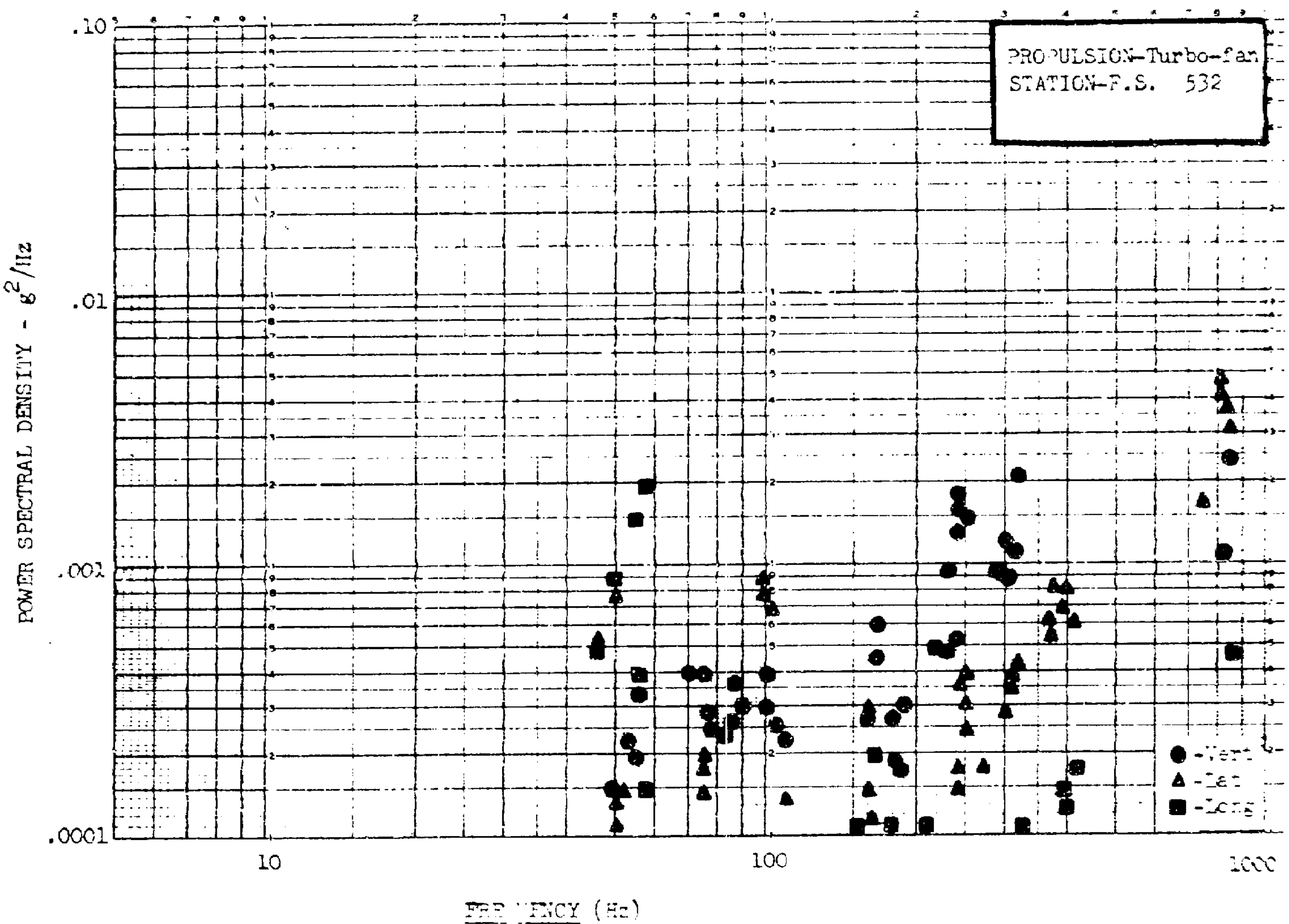

FIGURE 3-14 PSD PLOT FOR AIRCRAFT NO. 3

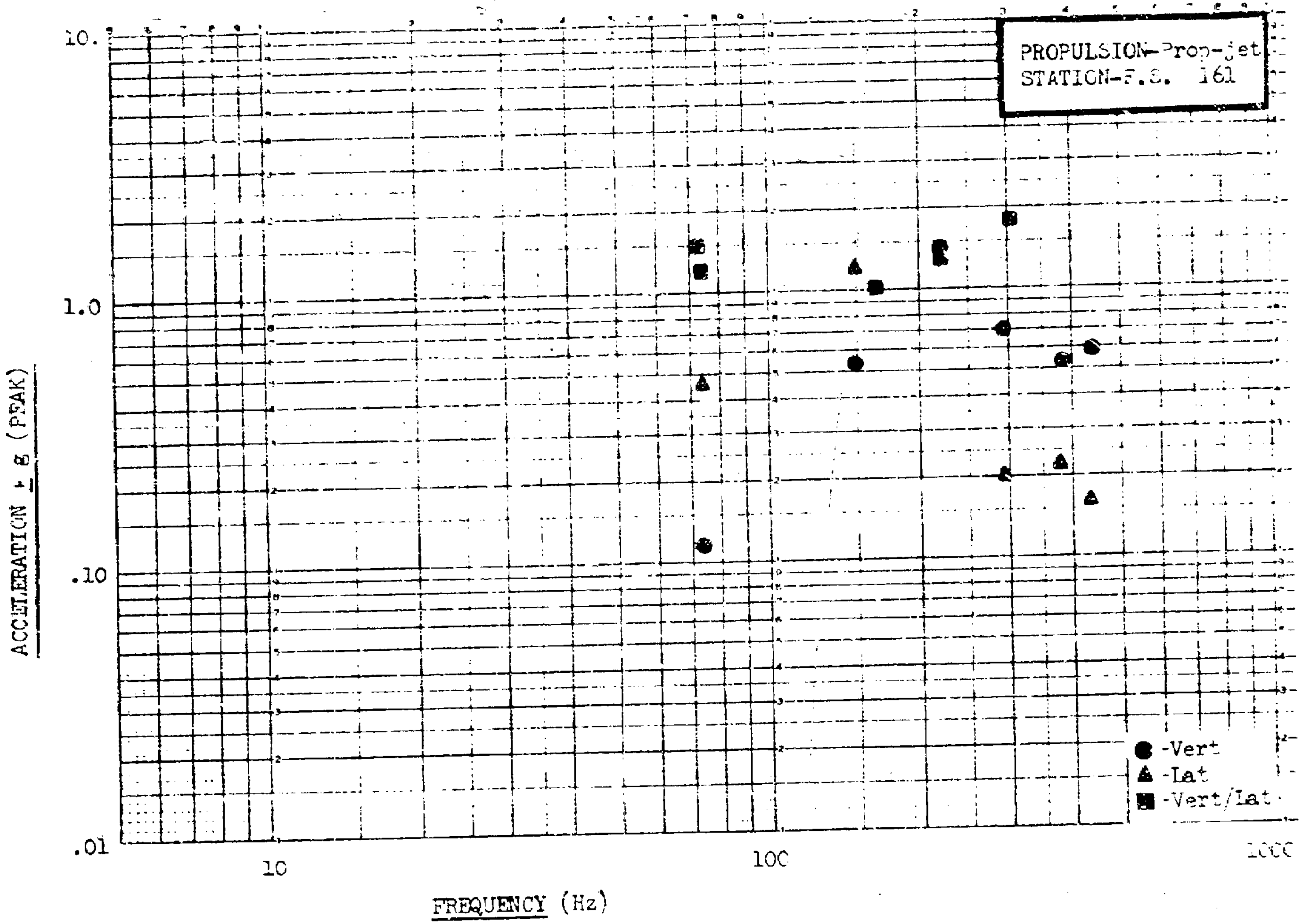

FIGURE C-15 ACCELERATION VERSUS FREQUENCY PLOT FOR AIRCRAFT NO. 4

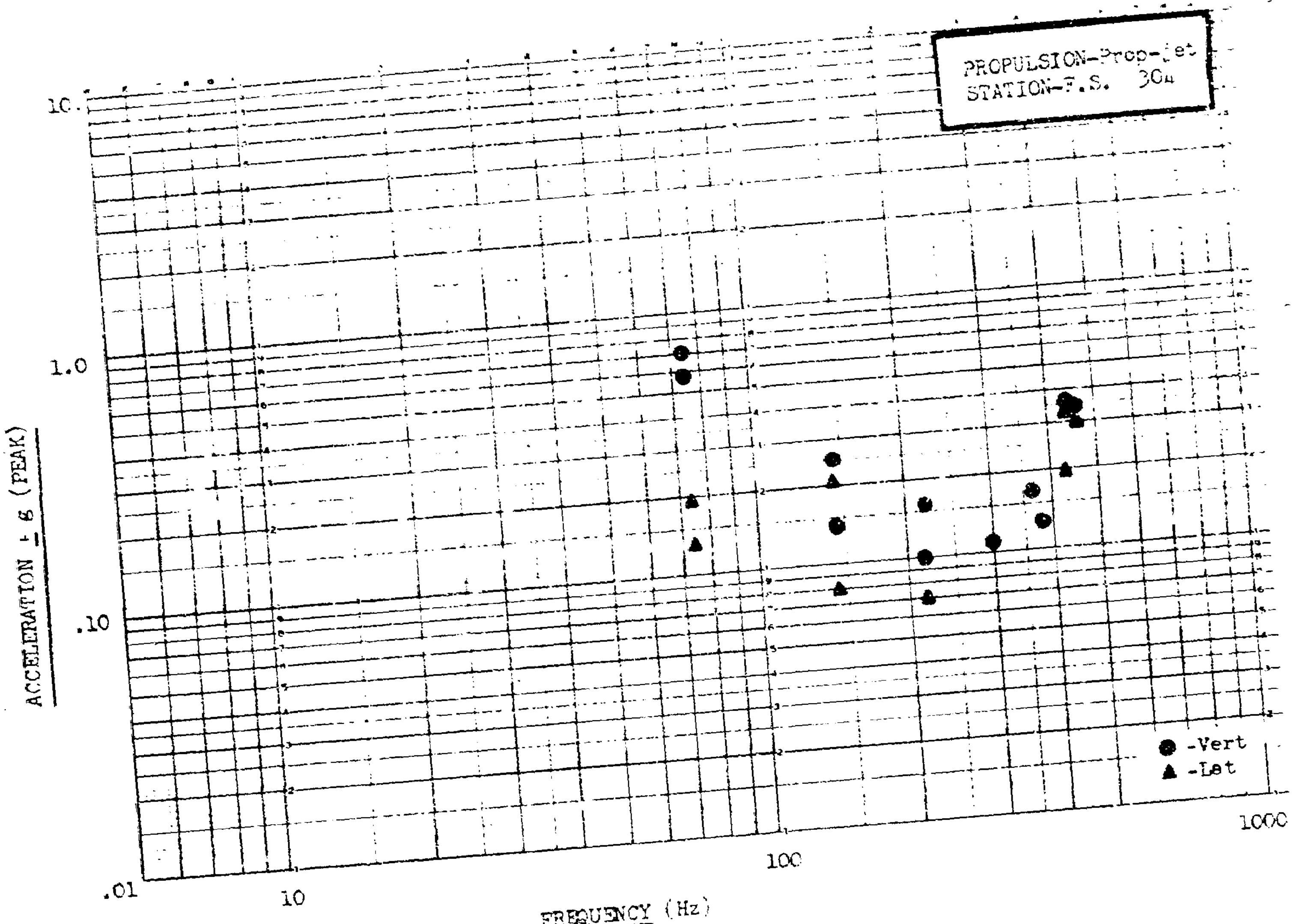

FIGURE C-16 ACCELERATION VERSUS FREQUENCY PLOT FOR AIRCRAFT NO. 4

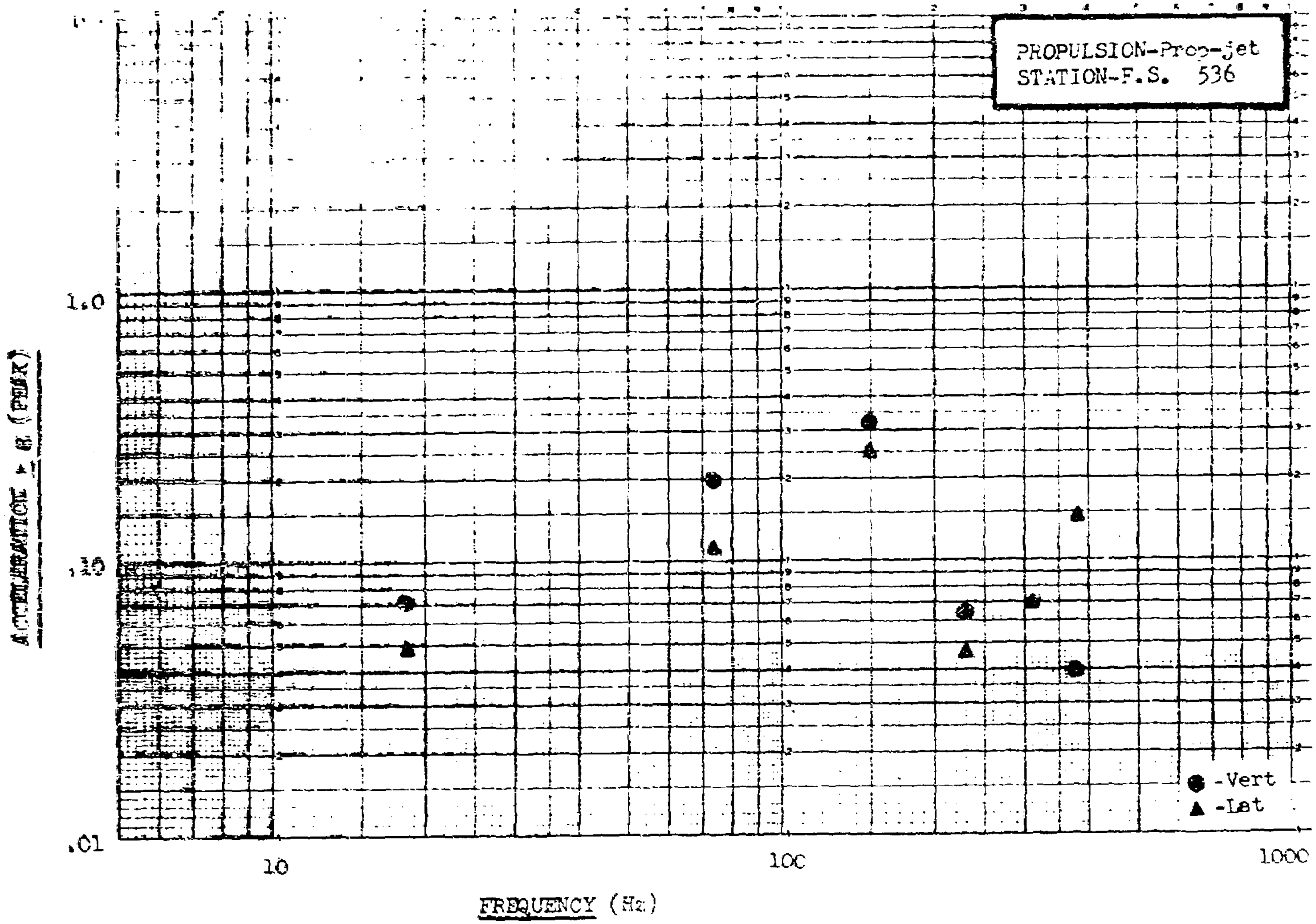

FIGURE C-17 ACCELERATION VERSUS FREQUENCY PLOT FOR AIRCRAFT NO. 4

TABLE C-1 - VIBRATION PARAMETERS

	TYPE		DURATION (hours)				TYPE		DURATION (Hours)		
WRA NO.	LAB	FIELD	LAB	FIELD	Δ	WRA NO.	LAB	FIELD	LAB	FIELD	Δ
1	Sine	Random	9	95	86	25	Sine	Random	9	70	61
2	Sine	Random	18	2690	2672	26	Sine	Sine	684	874	190
3	Sine	Random	70	400	330	27	Sine	Random	293	1270	977
4	Sine	Random	9	94	85	28	Sine	Random	9	94	85
5	Sine	Random	70	933	863	29	Sine	Sine	88	209	121
6	Sine	Random	9	85	76	30	Sine	Random	70	385	315
7	Sine	Random	70	375	305	31	Sine	Random	9	97	88
8	Sine	Random	18	1360	1342	32	Sine	Sine	684	874	190
9	Sine	Sine	88	209	121	33	Sine	Sine	88	209	121
10	Sine	Random	9	101	92	34	Sine	Sine	684	874	190
11	Sine	Random	167	478	311	35	Sine	Random	490	1130	640
12	Sine	Random	167	478	311	36	Sine	Sine	88	209	121
13	Sine	Random	70	410	340	37	Sine	Random	293	1470	1177
14	Sine	Sine	680	955	275	38	Sine	Random	300	814	514
15	Sine	Random	9	84	75	39	Sine	Random	300	814	514
16	Sine	Random	9	84	75	40	Sine	Random	300	814	514
17	Sine	Random	9	83	74	41	Sine	Sine	684	874	190
18	Sine	Random	9	84	75	42	Sine	Sine	684	874	190
19	Sine	Random	18	905	887	43	Sine	Sine	684	874	190
20	Sine	Random	70	2110	2040	44	Sine	Random	70	1830	1760
21	Sine	Sine	88	209	121	45	Sine	Sine	680	955	275
22	Sine	Random	70	1900	1830	46	Sine	Random	177	600	423
23	Sine	Sine	684	874	190	47	Sine	Sine	684	874	190
24	Sine	Random	161	787	626	48	Sine	Random	292	944	652

TABLE C-1 - VIBRATION PARAMETERS (Continued)

	TYPE		DURATION (Hours)				TYPE		DURATION (Hours)		
WRA NO.	LAB	FIELD	LAB	FIELD	Δ	WRA NO.	LAB	FIELD	LAB	FIELD	Δ
49	Sine	Sine	680	955	275	73	Sine	Random	70	485	415
50	Sine	Sine	9	79	70	74	Sine	Random	9	89	80
51	Sine	Sine	200	1160	960	75	Sine	Random	70	400	330
52	Sine	Sine	684	874	190	76	Sine	Sine	684	874	190
53	Sine	Sine	300	814	500	77	Sine	Random	9	84	75
54	Sine	Random	9	90	81	78	Sine	Sine	684	874	190
55	Sine	Random	293	1380	1087	79	Sine	Random	9	92	83
56	Sine	Random	9	84	75	80	Sine	Random	167	478	311
57	Sine	Random	300	814	500	81	Sine	Sine	684	874	190
58	Sine	Random	9	84	75	82	Sine	Random	70	378	308
59	Sine	Random	183	6320	6137	83	Sine	Random	9	84	75
60	Sine	Random	9	84	75	84	Sine	Sine	88	209	121
61	Sine	Random	292	944	652	85	Sine	Random	70	1900	1830
62	Sine	Random	9	84	75	86	Sine	Sine	684	874	190
63	Sine	Sine	88	209	121	87	Sine	Random	300	814	514
64	Sine	Random	9	84	75	88	Sine	Random	300	814	514
65	Sine	Random	292	944	652	89	Sine	Random	300	814	514
66	Sine	Random	70	1900	1830	90	Sine	Sine	684	874	190
67	Sine	Random	9	88	79	91	Sine	Random	300	814	514
68	Sine	Random	70	345	275	92	Sine	Sine	88	209	121
69	Sine	Sine	88	209	121	93	Sine	Sine	684	874	190
70	Sine	Random	9	96	87	94	Sine	Sine	88	209	121
71	Sine	Random	167	478	311	95	Sine	Sine	684	874	190
72	Sine	Random	9	84	75						

APPENDIX D

MTBF Data

The calculated MTBF's of all WRA's for both demonstration testing and field operations are summarized in the following Tables. These summaries reflect the results of the failure reclassification according to the ground rules and assumptions, described in Section IV. The reprediction of each WRA's reliability using the coordination copy of MIL-HDBK-217B and the Mean Flight Time Between Maintenance Actions (MFTBMA) are also tabularized.

TABLE D-1 DEMONSTRATED MTBF VALUES

WRA NUMBER	DEMONSTRATED MTBF BEFORE RECLASSIFYING	DEMONSTRATED MTBF AFTER RECLASSIFYING	WRA NUMBER	DEMONSTRATED MTBF BEFORE RECLASSIFYING	DEMONSTRATED MTBF AFTER RECLASSIFYING
1	381	381	22	22,222	15,385
2	164	164	23	3,861	645
3	212	148	24	1,054	526
4	361	361	25	2,898	2,898
5	2,506	1,736	26	3,279	546
6	226	226	27	417	334
7	665	466	28	618	618
8	164	164	29	312	187
9	421	252	30	1,969	1,376
10	226	226	31	2,203	2,203
11	3,378	1,206	32	7,813	1,307
12	7,299	2,611	33	198	80
13	497	348	34	66,667	10,870
14	1,527	508	35	5,102	3,540
15	38,462	38,462	36	1,745	1,047
16	38,462	38,462	37	347	278
17	743	743	38	9,434	9,434
18	38,462	38,462	39	8,696	8,696
19	612	612	40	10,638	10,638
20	4,184	2,898	41	55,556	9,346
21	442	265	42	62,500	10,417

TABLE D-1 DEMONSTRATED MTBF VALUES (Continued)

WRA NUMBER	DEMONSTRATED MTBF BEFORE RECLASSIFYING	DEMONSTRATED MTBF AFTER RECLASSIFYING	WRA NUMBER	DEMONSTRATED MTBF BEFORE RECLASSIFYING	DEMONSTRATED MTBF AFTER RECLASSIFYING
43	58,824	9,804	63	2,000	1,200
44	2,049	1,420	64	62,500	62,500
45	1,776	591	65	1,597	726
46	756	168	66	13,158	9,091
47	9,091	1,520	67	910	910
48	1,129	513	68	789	552
49	883	294	69	1,462	877
50	370	370	70	988	988
51	2,453	2,453	71	1,901	678
52	5,917	985	72	6,993	6,993
53	100,000	100,000	73	1,043	729
54	1,426	1,426	74	1,159	1,159
55	1,024	819	75	1,189	832
56	12,195	12,195	76	14,085	2,331
57	200,000	200,000	77	5,586	5,586
58	6,289	6,289	78	83,333	14,085
59	1,732	1,732	79	3,300	3,300
60	7,874	7,874	80	1,092	390
61	1,580	718	81	15,625	2,591
62	12,500	12,500	82	1,582	1,107

TABLE D-1 DEMONSTRATED MTBF VALUES (Continued)

WRA NUMBER	DEMONSTRATED MTBF BEFORE RECLASSIFYING	DEMONSTRATED MTBF AFTER RECLASSIFYING	WRA NUMBER	DEMONSTRATED MTBF BEFORE RECLASSIFYING	DEMONSTRATED MTBF AFTER RECLASSIFYING
83	4,184	4,184	90	76,923	12,346
84	405	243	91	50,000	50,000
85	10,753	7,463	92	125,000	71,428
86	20,833	3,521	93	27,777	4,673
87	16,949	16,949	94	125,000	71,428
88	16,949	16,949	95	14,706	2,457
89	11,236	11,236			

TABLE D-2 FIELD MTBF VALUES

WRA NUMBER	FIELD MTBF (MFTBMA)	FIELD MTBF AFTER RECLASSIFICATION	WRA NUMBER	FIELD MTBF (MFTBMA)	FIELD MTBF AFTER RECLASSIFICATION
1	111	193	21	2,153	1,141
2	297	60	22	424	646
3	30	141	23	135	292
4	66	113	24	89	253
5	343	404	25	330	2,048
6	65	128	26	308	511
7	76	296	27	73	229
8	120	31	28	35	85
9	227	163	29	154	88
10	51	87	30	122	780
11	200	2,800	31	387	646
12	667	5,386	32	1,076	2,045
13	72	331	33	917	847
14	431	954	34	2,153	2,045
15	12,850	17,344	35	76	591
16	4,452	12,022	36	3,107	1,653
17	65	126	37	138	595
18	12,850	17,344	38	333	1,467
19	194	82	39	238	1,833
20	131	63	40	175	917

TABLE D-2 FIELD MTBF VALUES (Continued)

WRA NUMBER	FIELD MTBF (MFTBMA)	FIELD MTBF AFTER RECLASSIFICATION	WRA NUMBER	FIELD MTBF (MFTBMA)	FIELD MTBF AFTER RECLASSIFICATION
41	615	1,364	61	85	287
42	2,153	5,902	62	278	401
43	861	4,091	63	3,107	1,653
44	43	30	64	424	572
45	431	477	65	98	347
46	10	36	66	318	143
47	718	1,636	67	56	94
48	152	926	68	72	235
49	308	636	69	718	382
50	28	89	70	56	96
51	51	196	71	175	667
52	239	256	72	312	491
53	1,667	7,333	73	98	330
54	283	437	74	76	133
55	298	454	75	59	309
56	495	707	76	215	227
57	238	917	77	618	220
58	234	401	78	2,153	2,045
59	134	45	79	59	129
60	312	616	80	133	778

TABLE D-2 FIELD MTBF VALUES (Continued)

WRA NUMBER	FIELD MTBF (MFTBMA)	FIELD MTBF AFTER RECLASSIFICATION	WRA NUMBER	FIELD MTBF (MFTBMA)	FIELD MTBF AFTER RECLASSIFICATION
81	8,612	8,181	89	3,333	10,579
82	136	796	90	215	341
83	101	291	91	4,810	10,579
84	269	571	92	3,107	1,653
85	524	430	93	3,107	2,950
86	4,306	5,902	94	3,107	1,653
87	3,333	10,579	95	1,076	2,045
88	1,667	7,333			

TABLE D-3 PREDICTED, DEMONSTRATED AND FIELD MTBF VALUES

WRA NUMBER	MTBF PRE-DICTION USING MIL-HDBK-217B	DEMONSTRATED MTBF AFTER RECLASSIFYING	FIELD MTBF (RECLASSIFIED)	WRA NUMBER	MTBF PRE-DICTION USING MIL-HDBK-217B	DEMONSTRATED MTBF AFTER RECLASSIFYING	FIELD MTBF (RECLASSIFIED)
1	274	381	193	21	6,006	265	1,141
2	290	164	60	22	1,097	15,385	646
3	175	148	141	23	1,067	645	292
4	279	361	113	24	262	526	253
5	140	1,736	404	25	399	2,898	2,048
6	171	226	128	26	1,630	546	511
7	730	466	296	27	744	334	229
8	290	164	31	28	778	618	85
9	1,377	252	163	29	602	187	88
10	163	226	87	30	1,374	1,376	780
11	3,076	1,206	2,800	31	556	2,203	646
12	6,901	2,611	5,386	32	2,345	1,307	2,045
13	434	348	331	33	5,625	80	847
14	5,555	508	954	34	12,837	10,870	2,045
15	19,415	38,462	17,344	35	945	3,540	591
16	115,929	38,462	12,022	36	4,815	1,047	1,653
17	872	743	126	37	443	278	595
18	52,596	38,462	17,344	38	607	9,434	1,467
19	899	612	82	39	557	8,696	1,833
20	430	2,898	63	40	671	10,638	917

TABLE D-3 PREDICTED, DEMONSTRATED AND FIELD MTBF VALUES (Continued)

WRA NUMBER	MTBF PREDICTION USING MIL-HDBK-217B	DEMONSTRATED MTBF AFTER RECLASSIFYING	FIELD MTBF (RECLASSIFIED)	WRA NUMBER	MTBF PREDICTION USING MIL-HDBK-217B	DEMONSTRATED MTBF AFTER RECLASSIFYING	FIELD MTBF (RECLASSIFIED)
41	10,906	9,346	1,364	61	1,928	718	287
42	12,846	10,417	5,902	62	5,076	12,500	401
43	13,356	9,804	4,091	63	12,243	1,200	1,653
44	176	1,420	30	64	68,961	62,500	572
45	5,841	591	477	65	1,887	726	347
46	80	168	36	66	1,720	9,091	143
47	1,483	1,520	1,636	67	969	910	94
48	314	513	926	68	458	552	235
49	1,609	294	636	69	2,003	877	382
50	205	370	89	70	1,157	988	96
51	2,295	2,453	196	71	653	678	667
52	748	985	256	72	2,412	6,993	491
53	5,487	100,000	7,333	73	2,756	729	330
54	3,129	1,426	437	74	883	1,159	133
55	2,656	819	454	75	2,776	832	309
56	6,987	12,195	707	76	2,700	2,331	227
57	12,898	200,000	917	77	2,468	5,586	220
58	3,093	6,289	401	78	19,105	14,085	2,045
59	581	1,732	45	79	840	3,300	129
60	10,677	7,874	616	80	433	390	778

TABLE D-3 PREDICTED, DEMONSTRATED AND FIELD MTBF VALUES (Continued)

WRA NUMBER	MTBF PREDICTION USING MIL-HDBK-217B	DEMONSTRATED MTBF AFTER RECLASSIFYING	FIELD MTBF (RECLASSIFIED)	WRA NUMBER	MTBF PREDICTION USING MIL-HDBK-217B	DEMONSTRATED RECLASSIFYING	FIELD MTBF (RECLASSIFIED)
81	5,858	2,591	8,181	89	11,745	11,236	10,579
82	1,128	1,107	796	90	2,924	12,346	341
83	2,544	4,184	291	91	585,823	50,000	10,579
84	617	243	571	92	2,450,980	71,428	1,653
85	1,404	7,463	430	93	70,482	4,673	2,950
86	8,416	3,521	5,902	94	1,400,560	71,428	1,653
87	19,686	16,949	10,579	95	14,286	2,457	2,045
88	19,043	16,949	7,333				

APPENDIX E

Parts and Burn-In Test Data

This appendix summarizes the basic parts and burn-in information for each WRA. The total number of peice parts and relative percentages of micro-circuits and high reliability parts for each WRA are shown. The results of the burn-in tests that were conducted on demonstration and production items are also included. The duration, average number of failures and percentage of functions monitored during these tests are tabularized.

TABLE E-1 PARTS QUALITY MEASURES

WRA NUMBER	PIECE PARTS (EXCLUDING MISCELLANEOUS HARDWARE)	% MICROCIRCUITS	% RESISTORS, CAPACITORS, & SEMICONDUCTORS TX OR BETTER	WRA NUMBER	PIECE PARTS (EXCLUDING MISCELLANEOUS HARDWARE)	% MICROCIRCUITS	% RESISTORS, CAPACITORS, & SEMICONDUCTORS TX OR BETTER
1	1531	8	10	20	1820	42	41
2	1806	0	15	21	633	3	61
3	2875	7	48	22	843	3	65
4	1834	6	10	23	2906	65	64
5	4338	3	56	24	2259	12	19
6	2469	8	12	25	804	13	18
7	643	2	26	26	1016	39	5
8	1806	0	16	27	1537	42	0
9	3307	6	56	28	975	64	31
10	2381	9	13	29	2048	23	65
11	491	0	78	30	629	26	36
12	238	0	79	31	1200	7	15
13	3549	12	34	32	1719	58	7
14	882	61	64	33	359	21	71
15	51	0	0	34	392	96	0
16	14	0	0	35	303	0	0
17	1292	15	14	36	497	16	51
18	33	0	0	37	4390	36	0
19	989	5	31	38	2010	16	3

TABLE E-1 PARTS QUALITY MEASURES (Continued)

WRA NUMBER	PIECE PARTS (EXCLUDING MISCELLANEOUS HARDWARE)	% MICROCIRCUITS	% RESISTORS CAPACITORS, & SEMICONDUCTORS TX OR BETTER	WRA NUMBER	PIECE PARTS (EXCLUDING MISCELLANEOUS HARDWARE)	% MICROCIRCUITS	% RESISTORS CAPACITORS, & SEMICONDUCTORS TX OR BETTER
39	2363	14	3	58	63	0	2
40	1797	12	6	59	1963	11	36
41	519	97	0	60	56	4	3
42	444	96	0	61	634	9	33
43	450	93	0	62	59	0	0
44	5823	9	18	63	133	0	56
45	755	67	69	64	17	0	0
46	2311	29	46	65	549	9	39
47	1427	15	2	66	605	0	8
48	1851	51	25	67	790	22	26
49	701	53	52	68	535	21	56
50	6678	41	0	69	842	4	36
51	1345	26	79	70	1041	21	5
52	778	17	12	71	1460	37	62
53	31	0	0	72	150	0	45
54	628	28	20	73	591	2	51
55	233	0	0	74	447	2	28
56	45	0	0	75	565	3	50
57	28	7	0	76	616	12	2

TABLE E-1 PARTS QUALITY MEASURES (Continued)

WRA NUMBER	PIECE PARTS (EXCLUDING MISCELLANEOUS HARDWARE)	% MICROCIRCUITS	% RESISTORS CAPACITORS, & SEMICONDUCTORS TX OR BETTER	WRA NUMBER	PEICE PARTS (EXCLUDING MISCELLANEOUS HARDWARE)	% MICROCIRCUITS	% RESISTORS CAPACITORS, & SEMICONDUCTORS TX OR BETTER
77	55	0	67	87	85	7	0
78	148	0	0	88	86	8	0
79	655	0	29	89	143	6	0
80	938	7	67	90	902	39	0
81	481	0	11	91	3	0	0
82	750	1	39	92	4	0	0
83	237	1	29	93	52	0	0
84	1748	4	64	94	7	0	0
85	144	3	64	95	45	0	0
86	278	4	19				

TABLE E-2 BURN-IN TEST PARAMETERS AND TEST EFFICIENCY

WRA NUMBER	DURATION (HRS) DEMONSTRATION	PRODUCTION UNITS	AVERAGE # FAILURES PRODUCTION UNITS	% FUNCTIONS MONITORED
1	44	40	1.35	80
2	55	40	2.00	80
3	350	45	1.76	10
4	44	40	0.90	80
5	35	40	2.50	80
6	44	40	2.32	80
7	350	45	0.79	10
8	55	40	0	80
9	150	125	1.35	98
10	44	40	2.05	80
11	350	250	0	80
12	350	250	0	80
13	350	45	0.47	10
14	150	125	0.91	90
15	44	40	0.10	80
16	44	40	0.05	80
17	44	40	1.32	80
18	44	40	0	80
19	55	40	0.52	80
20	35	40	0.50	80
21	150	125	0.35	98
22	35	40	0	80
23	150	125	0	100
24	500	400	0.78	95
25	44	40	0.125	80
26	150	125	0.125	100
27	130	100	0.20	90
28	44	40	1.05	80
29	150	125	0.04	98

TABLE E-2 BURN-IN TEST PARAMETERS AND TEST EFFICIENCY (Con't)

WRA NUMBER	DURATION (HRS)		AVERAGE # FAILURES PRODUCTION UNITS	% FUNCTIONS MONITORED
	DEMONSTRATION	PRODUCTION UNITS		
30	350	45	0.10	10
31	44	40	0.225	80
32	150	125	0.625	100
33	150	125	1.02	98
34	150	125	0	100
35	44	40	1.00	80
36	150	125	0.26	98
37	130	100	0.55	90
38	350	250	0.89	50
39	350	250	0.72	50
40	350	250	0.42	50
41	150	125	0	100
42	150	125	0	100
43	150	125	0	100
44	35	4	2.50	80
45	150	125	0.54	90
46	215	190	9.90	80
47	150	125	0.67	100
48	500	190	0.88	90
49	150	125	0.83	90
50	44	40	5.33	80
51	150	125	0	75
52	150	125	0	100
53	350	250	0.02	50
54	44	40	0.55	80
55	130	100	0.15	90
56	44	40	0.025	80
57	350	250	0.094	50
58	44	40	0.125	80

TABLE E-2 BURN-IN TEST PARAMETERS AND TEST EFFICIENCY (Con't)

WRA NUMBER	DURATION (HRS) DEMONSTRATION	PRODUCTION UNITS	AVERAGE # FAILURES PRODUCTION UNITS	% FUNCTIONS MONITORED
59	0	100	0.033	50
60	44	40	0.15	80
61	500	190	0.25	90
62	44	40	0.05	80
63	150	125	0.87	98
64	44	40	0.025	80
65	500	190	0.18	90
66	35	40	1.00	80
67	44	40	1.77	80
68	350	45	0.61	10
69	150	125	1.26	98
70	44	40	0.40	80
71	350	250	4.50	80
72	44	40	0.35	80
73	50	45	0.47	10
74	44	40	0.70	80
75	350	45	0.24	10
76	150	125	0.125	100
77	44	40	0.65	80
78	150	125	0	100
79	44	40	0.25	80
80	350	250	3.50	80
81	150	125	0.125	100
82	350	45	0.11	10
83	44	40	0.175	80
84	150	125	1.83	98
85	35	40	0.50	80
86	150	125	0	100
87	350	250	0	50

TABLE E-2 BURN-IN TEST PARAMETERS AND TEST EFFICIENCY (Con't)

WRA NUMBER	DURATION (HRS) DEMONSTRATION	PRODUCTION UNITS	AVERAGE # FAILURES PRODUCTION UNITS	% FUNCTIONS MONITORED
88	350	250	0.075	50
89	350	250	0.075	50
90	150	125	0	100
91	350	250	0.13	50
92	150	125	0	98
93	150	125	0	100
94	150	125	0.22	98
95	150	125	0.125	100